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RDT&E PROJECT NO.
USATECOM PROJECT NO. 4-5-1591-01
USAAVNTA PROJECT NO. 65-12

ENGINEERING FLIGHT TEST OF UH-1B/540 ROTOR HELICOPTER EQUIPPED WITH XM-16/M-5, XM-21/M-5 OR XM-3/M-5 ARMAMENT SUBSYSTEM

FINAL REPORT

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PROJECT PILOTS

DECEMBER 1966

U. S. ARMY AVIATION TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA

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TEST REPORT

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ABSTRACT

This report presents the results of an engineering flight test of the UH-1B/540 rotor helicopter equipped with the XM-16, XM-21, or XM-3 armament subsystem in conjunction with the M-5 armament subsystem. The test was conducted by the U. S. Army Aviation Test Activity (USAAVNTA). Overall test objectives were to verify safety of flight, develop data for the operator's manual, and assure that aircraft modifications did not degrade the handling qualities or limit the operational characteristics of the subsystems. Specific objectives were to determine quantitatively the effect of the armament subsystems on stability, control and performance of the aircraft, to determine the rocket launcher jettison characteristics, and to define the usable limits of the flight envelope for safe jettison of the launchers. Testing was conducted at Edwards Air Force Base, California and at sites in Fort Irwin and Bakersfield, California. A total of 152 flights for a productive flight time of 116.4 hours was flown on aircraft S/N 64-14105 between 13 November 1965 and 5 May 1966. This included 30 jettison flights and 35 firing flights.

There were no significant adverse changes in the stability and control characteristics of the UH-1B/540 helicopter due to the installation of the various armament subsystems. A drag penalty imposed by the installation of the XM-3/M-5 or XM-21/M-5 caused a 13-percent and 10-percent decrease in specific range respectively with a corresponding 20-percent and 11-percent decrease in airspeed. The vibration level of the aircraft was generally satisfactory. A self-excited undamped lateral 2/3-per-rev vibration grounded the aircraft and terminated further testing on the XM-3/M-5 during the 9500 pounds stability and control portion of the test program. However, sufficient data was obtained prior to the termination of testing to indicate no problems will exist at the heavier gross weights. This characteristic could have safety-of-flight implications and should be corrected. Firing the various armament subsystems could be conducted at all airspeeds within the flight envelope with no major stability and control problem encountered. Firing rockets in a hover, with the launcher at negative deflection, should be avoided. Rocket launcher jettison can be satisfactorily accomplished under all level flight conditions tested. Jettison should be avoided during autorotations and in close proximity to the ground.

FOREWORD

The U. S. Army Test and Evaluation Command (USATECOM) assigned to the U. S. Army Aviation Test Activity (USAAVNTA) responsibility for preparing test plan, conducting test, and submitting final report.

PHOTO 1 - UH-1B/540 Helicopter

SECTION 1. INTRODUCTION

1.1 BACKGROUND

On 12 August 1964, the Iroquois Project Manager requested that USATECOM conduct confirmatory, airworthiness, performance and logistical tests of the UH-1B helicopter equipped with the Model 540 rotor system. A test program coordination meeting was held on 19 May 1965 and a decision was made to conduct separate tests for each armament subsystem to be installed on the UH-1B/540 helicopter.

USATECOM issued a test directive to USAAVNTA, on 16 July 1965, to conduct an engineering (product improvement) test of the UH-1B/540 rotor helicopter equipped with armament subsystems. USAAVNTA submitted a test plan which, as modified by reference b, was approved by USATECOM on 7 December 1965.

This test was conducted at Edwards Air Force Base, California, and at sites in Fort Irwin and Bakersfield, California. A total of 152 flights for a productive flight time of 116.4 hours was flown from 13 November 1965 through 5 May 1966.

Interim reports were submitted in references j, k, l and m, on 15 December 1965, 21 December 1965, and 28 March 1965 respectively.

1.2 DESCRIPTION OF MATERIEL

1.2.1 UH-1B/540 Helicopter

The UH-1B/540 helicopter is a utility helicopter powered by the T53-L-11, 1100 shaft horsepower gas turbine engine. Various weapon subsystems can be installed on this helicopter. The 540 rotor is a two-bladed teetering, semi-rigid rotor with a 44-foot diameter and 27-inch chord. The fuselage is identical to that of a standard UH-1B helicopter except for the addition of a UH-1D synchronizing elevator and a cambered vertical stabilizer. For a more detailed description see appendix III.

1.2.2 M-5 Armament Subsystem

The M-5 armament subsystem (figure A) is designed as a permanent installation on the UH-1B helicopter. The M-5 houses a 40-millimeter (mm) M-75 grenade launcher, which is an air-cooled, electric-motor driven, rapid-fire weapon capable of launching antipersonnel, fragmentation-type projectiles. The launcher is percussion fired and fed by a metallic belt. For a complete description see appendix III.

FIGURE A
The M-5 Armament Subsystem, a 40mm M-75 Grenade Launcher

1.2.3 XM-3 Armament Subsystem

The XM-3 armament subsystem (figure B) provides the capability of firing 2.75-inch limited-spin folding-fin aerial rockets (LSFFAR's) from a launcher mounted on either side of the helicopter. The launcher is of the open-breech tube type. Each launcher consists of 4 modules containing 6 tubes each.

The launcher has manual, mechanical adjustments from +6 degrees to -6 degrees relative to the waterline of the helicopter. The launcher can be jettisoned by means of explosive bolts in an emergency. For a complete description see appendix III.

FIGURE B
The XM-3 Armament Subsystem, a 2.75-inch Aerial Rocket

1.2.4 XM-16 Armament Subsystem

The XM-16 armament subsystem (figure C) is a combination of two 7-round 2.75-inch LSFFAR XM-157 rocket launchers and an M-6 subsystem consisting of four flexible 7.62-mm M-60C machine guns suspended from the universal stores mounts. The rocket launchers can be jettisoned by means of a solenoid actuated hook in an emergency. For a complete description see appendix III.

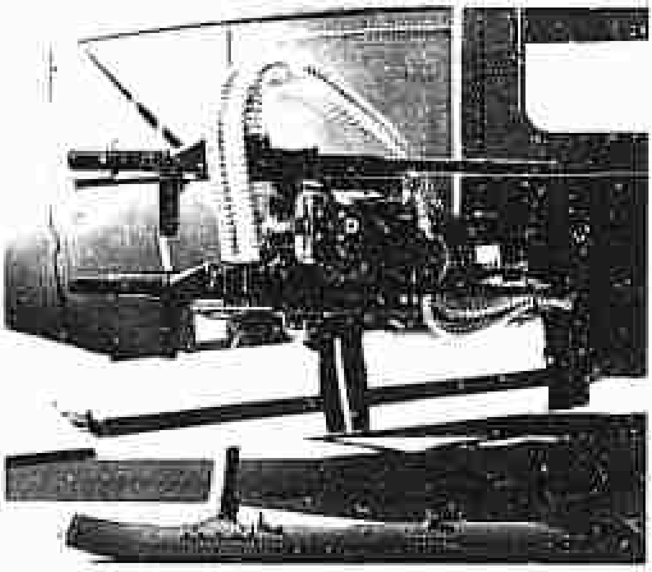

FIGURE C
The XM-16 Armament Subsystem Installed

1.2.5 XM-21 Armament Subsystem

The XM-21 armament subsystem (figure D) is identical to the XM-16 subsystem except that the M-6 subsystem is replaced by the XM-20 subsystem, which consists of two flexible 7.62-mm XM-134 mini-guns. The XM-134 mini-gun is a lightweight, electricity-driven, air-cooled, 6-barrel weapon capable of firing 4000 rounds per minute. For a complete description see appendix III.

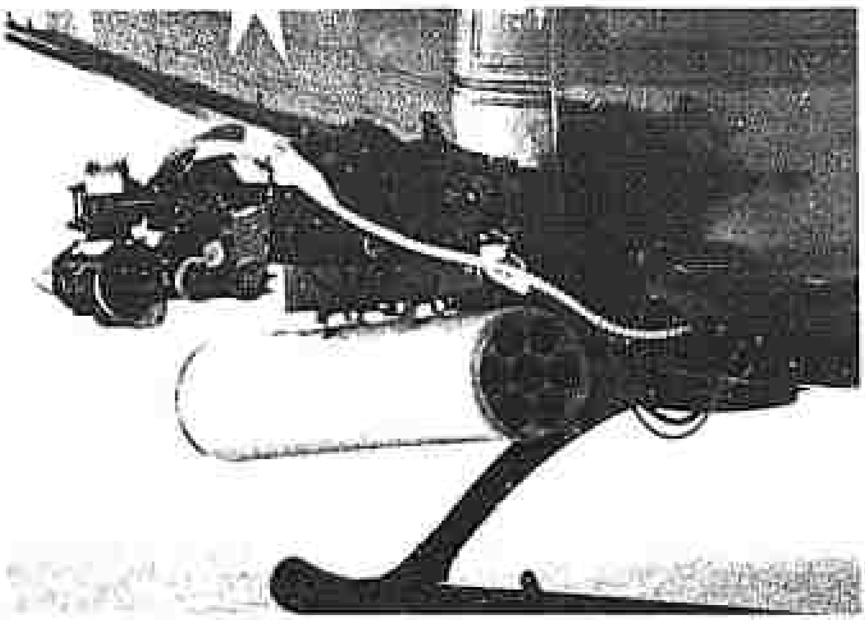

FIGURE D
The XM-21 Armament Subsystem Installed

1.2.6 Sights

The Mark 8 sight was used by the pilot during firing of the 2.75-inch rockets. The Mark 8 sight is obsolete and is being replaced by the M-60 sight.

The copilot/gunner used a prototype M-5/M-6 combination sight that allowed individual firing of the M-75 grenade launcher, M-60C machine gun or XM-134 mini-gun.

1.3 TEST OBJECTIVES

The overall objectives of this program were to verify safety of flight for the UH-1B/540 rotor helicopter equipped with each armament subsystem, develop data for inclusion in the aircraft operator's manual, and assure that the modifications made to the aircraft did not result in degradation of handling qualities or impose limitations on the operational characteristics of the armament subsystem.

The specific objectives of USAAVNTA in this test program were:

a. To determine the quantitative effect of the various armament subsystems on stability, control and performance of the aircraft.

b. To determine the rocket launcher jettison characteristics and define the usable limits of the flight envelope for the safe jettison of the launchers.

1.4 SUMMARY OF RESULTS

The significant results of this test are summarized below:

a. An unsatisfactory self-excited, undamped lateral 2/3-per-rev vibration developed that resulted in the termination of testing. This vibration could affect component life and, therefore, have safety-of-flight implications.

b. Firing from a hover with rocket launcher at maximum deflection was hazardous.

c. Compared with the unarmed UH-1B/540 test results (reference p), the installation of the XM-3/M-5 and XM-21/M-5 armament subsystems caused a reduction in specific range and airspeed of approximately 11 and 15 percent respectively.

d. The helicopter had negative dihedral effects at lightweight configurations and airspeeds above 50 knots calibrated airspeed (KCAS).

e. At a forward center of gravity (C.G.) there was insufficient longitudinal control to hover downwind at wind speeds greater than 20 knots.

f. During high-powered climbs in the speed range for maximum rate of climb (45 to 60 KCAS) the helicopter was dynamically unstable.

g. During firing of the XM-21 subsystem the XM-134 minigun failed to cease firing at the maximum inboard azimuth.

h. During firing of the M-5 subsystem with the combination sight the M-5 would not fire when the sight was rotated to the maximum azimuth.

i. When the XM-3 rocket launcher was jettisoned during autorotations the launcher floated on the skid and aft crosstube approximately 0.4 seconds before falling clear of the helicopter.

j. With protective shield installed on the leading edge of the synchronizing elevator there was no damage to the elevator.

For a more complete discussion of all results see section 2, Details of Test.

1.5 CONCLUSIONS

The performance, stability and control, and jettison characteristics of the UH-1B/540 helicopter equipped with the M-5, XM-16, XM-21, or XM-3 armament subsystem are considered to be satisfactory with the following exceptions:

a. A self-excited lateral 2/3-per-rev vibration of unknown origin was present that could affect component life and, therefore, have safety-of-flight implications (paragraph 2.4).

b. Firing from a hover with the rocket launcher at maximum deflection could create a hazardous situation with possible damage to the aircraft (paragraph 7.5).

c. At a forward C. G. there was insufficient longitudinal control remaining to hover downwind at wind speeds greater than 20 knots (paragraph 2.3.4).

d. During climbs in the speed range for maximum rate of climb (45 to 60 KCAS) the helicopter was dynamically unstable (paragraph 2.3.5).

e. The XM-134 mini-gun (XM-21 subsystem) when deflected to the maximum inboard azimuth failed to cease firing (paragraph 2.7.3).

f. The M-5 subsystem failed to fire when the combination sight was rotated to its maximum azimuth (paragraph 2.7.1).

g. The XM-3 rocket launcher contacted the skid and aft crosstube while jettisoning during autorotation (paragraph 2.6).

1.6 RECOMMENDATIONS

a. The contractor should define and correct the undamped lateral vibration prior to service test (paragraph 2.4).

b. The rocket launchers should not be at maximum deflection during firing in a hover (paragraph 2.7.5).

c. The operator's manual should include a warning that states: "Downwind approaches should be avoided and hovering downwind should not be attempted at wind speeds above 20 knots" (paragraph 2.3.4).

d. The airspeed for maximum performance climbs should be increased approximately 10 KCAS (paragraph 2.3.5).

e. The appropriate USATECOM test agency should investigate the failure of the XM-134 mini-gun to cease firing when deflected to the maximum inboard azimuth (paragraph 2.7.3).

f. The appropriate USATECOM test agency should investigate the M-5 combination sight compatibility (paragraph 2.7.1).

g. The operator's manual should include "caution" that states: "Jettison of the XM-3 rocket launcher should be avoided during autorotation. If jettison is necessary, accomplish at 60 KCAS and zero sideslip" (paragraph 2.6).

h. The level flight performance data in this report should be included in the operator's manual of the UH-1B/540 (paragraph 2.2.1).

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SECTION 2. DETAILS OF TEST

2.1 INTRODUCTION

Details of test methods and data reduction procedures corresponding to each test conducted may be found in appendix II. A brief description of the test methods used is included when necessary for clarification of the individual test. All tests were conducted with the XM-3, XM-16 or XM-21 armament subsystem installed. The M-5 armament subsystem was a permanent installation during the test program.

2.2 PERFORMANCE

2.2.1 Level Flight

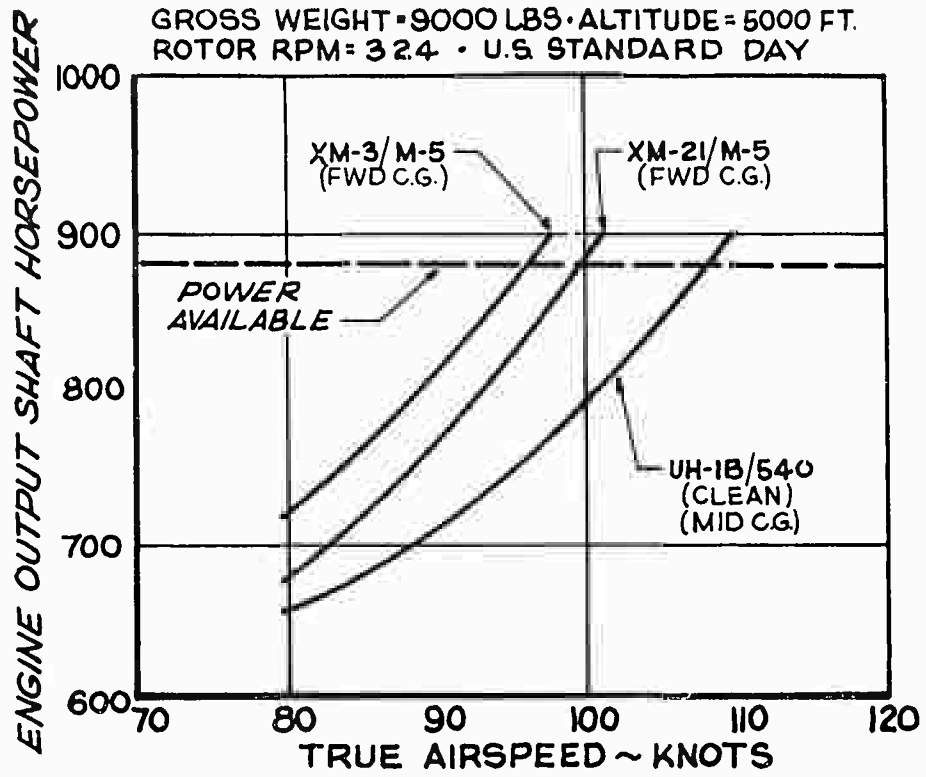
Level flight performance tests were conducted at gross weights ranging from 7570 pounds to 9215 pounds, rotor speed of 324 rpm, and density altitudes ranging from 4360 feet to 9060 feet. All tests were conducted at an average forward C.G. (126.3) with empty rocket launchers set at an elevation of 6 degrees above the waterline of the helicopter. Level flight performance tests with the XM-16 subsystem installed were not conducted since previous tests of the UH-1B showed less than 5-percent difference in power required between the XM-16 and XM-21 subsystems. The results of the individual tests are presented in figures 7 throug 13, appendix I, and summarized in non-dimensional form in figures 1 through 6.

Figure E shows a comparison of the power required in the armed and unarmed UH-1B/540 at 9000 pounds, 324 rotor rpm and 5000 feet. Data for the unarmed helicopter were obtained from reference p. The comparison was presented at different C.G. conditions since the normal loading configuration for a 9000-pound gross weight generally results in a forward C.G. for the armed aircraft and a near mid C.G. for the clean aircraft.

The reduction in specific range and airspeed caused by the installation of the XM-3/M-5 subsystems showed a maximum decrease

FIGURE E ⟶

Figure E



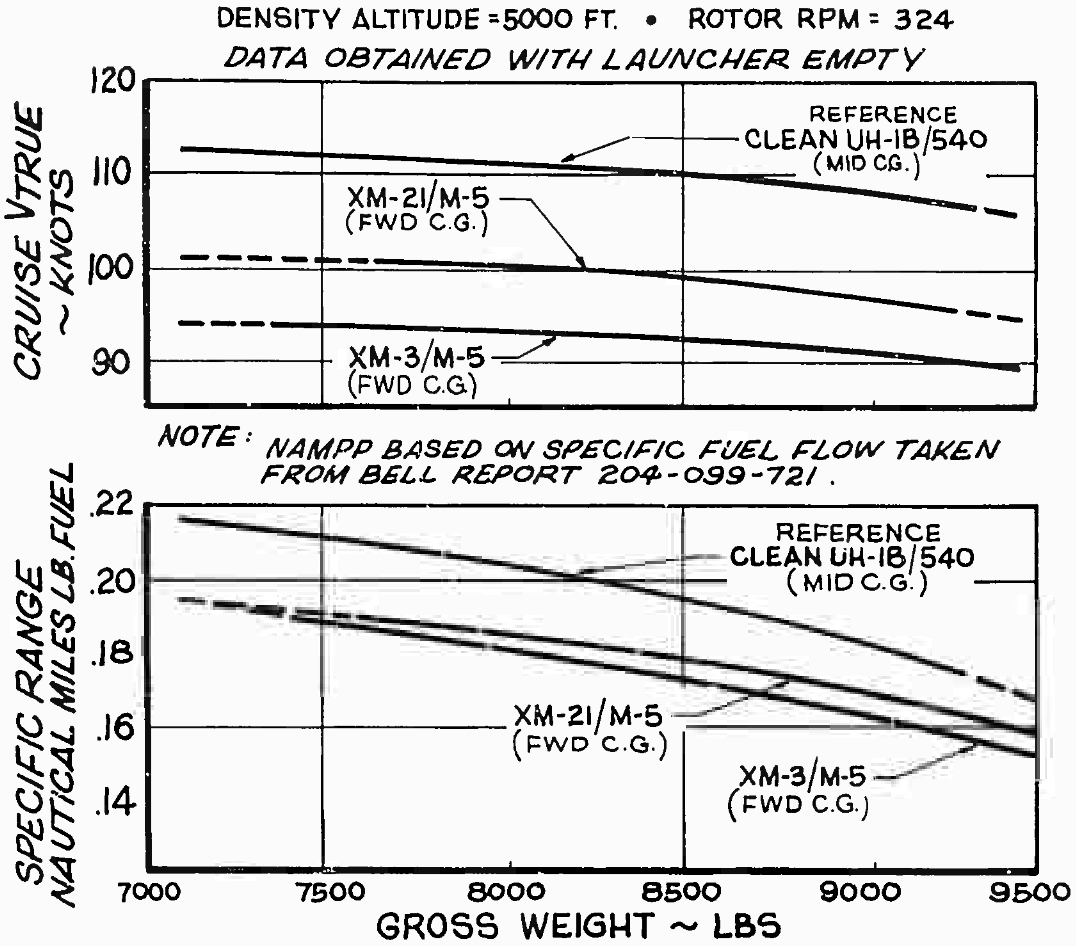
of 13 percent in specific range with a corresponding 20-percent decrease in airspeed (figure F). With the XM-21/M-5 subsystems installed the maximum decrease in specific range was 10 percent with a corresponding 11-percent decrease in airspeed (figure F). The radius of action for a clean aircraft was 117 nautical miles at 110 knots true airspeed (KTAS) compared with 103 nautical miles at 90 KTAS for the XM-21/M-5 configuration and 107 nautical miles at 96 KTAS for the XM-21/M-5 configuration. The radius of action was based on the following mission assumptions:

a. Engine-start gross weight of 9500 pounds with 1573 pounds of fuel aboard.

b. Warmup, takeoff and climb to cruise altitude (5000 feet) of 5 minutes at normal rated power.

c. Cruise at recommended airspeed (.99 Maximum Nautical Air Mile Per Pound of Fuel) at 5000 feet.

Figure F LEVEL FLIGHT RANGE SUMMARY



d. Loiter for 10 minutes at destination at normal rated power.

e. Return at recommended cruise airspeed at 5000 feet.

f. Landing at takeoff point with 10-percent usable fuel remaining (no distance allowance or fuel used assumed).

The reduction in cruise airspeed, specific range, and radius of action of the armed UH-1B/540 helicopter should be included in the UH-1B/540 operator's manual.

2.2.2 Autorotation

Autorotation tests were conducted at various airspeeds to determine the rate-of-descent characteristics as a function of airspeed and weight. The tests were conducted with the rocket launcher empty at an average density altitude of 5000 feet, 324 rotor rpm, forward C.G., and two average gross weights of 7800 pounds and 9330 pounds. At these conditions the armed UH-1B/540 had a minimum rate of descent of 1975 feet per minute (fpm) at 57.5 KCAS, as shown in figure 16, appendix I. The unarmed aircraft had a minimum rate of descent of 1780 fpm at 58 KCAS (reference p).

2.3 STABILITY AND CONTROL

The stability and control characteristics of the UH-1B/540 were evaluated at a gross weight range from 7500 pounds to 9300 pounds, rotor speed of 324 rpm, density altitude of 5000 feet, and forward C.G. (126.5). The characteristics were found to be essentially the same as those of the unarmed UH-1B/540 as reported in reference p.

2.3.1 Control Positions in Level Flight

Trimmed control positions as a function of calibrated airspeed were obtained during level flight performance tests. Control positions were satisfactory throughout the level flight envelope with sufficient control margin remaining at the maximum level flight airspeed. It was possible to trim all control forces to zero throughout the speed range tested and the control system exhibited positive self-centering characteristics. The results of the individual tests are presented in figures 17 through 23, appendix I.

2.3.2 Static Longitudinal Stability

The collective-fixed technique was used to evaluate static longitudinal stability. The aircraft was stabilized at a trim airspeed for the particular flight condition, the collective was locked in this position, then the airspeed was increased and/or decreased from this trim value by displacing the longitudinal cyclic control. Tests were conducted in level flight with empty rocket launchers.

The armed UH-1B/540 had satisfactory positive static longitudinal stability at all trim speeds above 40 KCAS. When the aircraft was trimmed at 39 KCAS, the static longitudinal stability between 14 and 30 KCAS was negative. This characteristic, which is inherent in all UH-1 helicopters, did not appear to be magnified with the weapon subsystem installed. This region of negative stability was within the limits of paragraph 3.2.10 of MIL-H-8501A and qualitatively was not considered objectionable. The results of the individual tests are presented in figures 24 through 26, appendix I.

2.3.3 Static Lateral-Directional Stability

Static lateral-directional stability was evaluated in level flight at trim speeds of 55, 83, and 100 KCAS; and in a powered descent at limit airspeed (V_{NE}). Static lateral-directional stability was also investigated in climbs at best climb airspeed and in autorotations at the airspeed for minimum rate of descent.

Installation of the various armament subsystems did not alter the positive static directional stability of the helicopter. The static directional stability became stronger with an increase in airspeed. The pedal deflection with increasing sideslip angles was essentially linear. The strong static directional stability permits pedal-fixed turns and enables maneuvering or turning to be accomplished with very little pedal application.

The change in static lateral-directional stability (dihedral effect) appeared to be a function of gross weight. At approximately 7700 pounds the dihedral effect was negative in level flight at airspeeds above 50 KCAS. At approximately 9200 pounds the dihedral effect was neutral to slightly positive in level flight from 56 KCAS to 115 KCAS. The dihedral effect during best rate of climb and minimum rate of descent airspeeds was neutral to slightly positive for all gross weights. The negative dihedral effect was also apparent in the unarmed helicopter. This characteristic, although not in accordance with paragraph 3.3.9 of MIL-H-8501A, was not considered objectionable by the pilot. Qualitative results of the static lateral-directional tests are presented in figures 27 through 47, appendix I.

2.3.4 Sideward and Rearward Flight

Sideward and rearward flight characteristics were evaluated in ground effect (IGE) at a forward C.G. and 8300 and 8100 pounds. These tests were conducted with the XM-16/M-5 armament subsystems installed. Results of these tests are shown in figures 48 and 49, appendix I.

The armed UH-1B/540 had acceptable sideward flight characteristics. Control positions as a function of sideward velocity were comparable with the unarmed UH-1B/540 test results but differed from the classical control positions in sideward flight. In left sideward flight the lateral control position remained essentially unchanged approximately 18 knots, then required an increase in left stick. During left sideward flight a sharp increase in right pedal was required between 9 to 17 KTAS. The helicopter exhibited a strong nosedown pitch in left sideward flight requiring aft cyclic control to maintain a near constant pitch attitude. Although the control positions were abnormal, the characteristics were not objectionable. Control positions during right sideward flight were essentially linear.

Rearward flight was conducted IGE at 8100 pounds and a C.G. of 127.5 inches (1.5 inches from full forward limit). As shown in figure 49, appendix I, an abrupt nosedown pitch change occurred between 5 and 15 KTAS that required approximately 3 inches of aft longitudinal control to maintain a constant pitch attitude. This characteristic was also observed in the unarmed UH-1B/540 (reference p). At 30 KTAS rearward flight 0.57 inches of aft control remained. A warning should be placed in the operator's manual that states: "Downwind approaches should be avoided and hovering 'downwind' should not be attempted at wind speeds above 20 knots."

2.3.5 Dynamic Stability

Dynamic stability tests were conducted during level flight, climbs, and autorotations. In level flight and autorotations the response of the helicopter to a 1-inch control pulse was essentially deadbeat about all three axes. The dutch roll mode excited by either a lateral or directional disturbance was well damped.

During steady-state climbs at maximum performance climb speed (45 to 60 KCAS), a slow divergent pitch oscillation occured. A longitudinal disturbance aggravated this instability which when not corrected immediately resulted in a significant loss in altitude. Constant monitoring of the pitch attitude was required during climbs in this speed range. This characteristic has been observed in all UH-1 series aircraft in varying degrees. To avoid this condition, climbs should be conducted approximately 10 KCAS above the maximum performance climb speed.

2.3.6 Controllability

The controllability of the aircraft with the armament subsystems installed was determined by noting the aircraft's response to step-type control inputs about all three axes. Representative controllability curves are presented in figures 56 through 67, appendix I, and summarized in figures 50 through 55.

The comparisons in tables I and II show that the controllability of the armed aircraft was essentially the same as that of the clean aircraft. The controllability characteristics in the armed configuration were considered satisfactory.

TABLE I

CONTROL SENSITIVITY

Configuration	Axis	Sensitivity deg/sec^2/in		Time to Peak sec
Clean	Pitch	Up 12.2	Dn 13.2	0.4 to 0.6
	Roll	Rt 22.4	Lt 20.5	0.3 to 0.5
	Yaw	Rt 35.8	Lt 31.5	0.4 to 0.5
Armed	Pitch	Up 14.0	Dn 14.5	0.5
	Roll	Rt 22.2	Lt 22.2	0.45
	Yaw	Rt 30	Lt 28.5	0.5

NOTE: Comparison made at 95 KCAS and 9300 pounds gross weight.

TABLE II

CONTROL RESPONSE

Configuration	Axis	Response deg/sec/in		Time to Peak sec
Clean	Pitch	Up 3.5	Dn 2.3	1.0 to 2.0
	Roll	Rt 9.8	Lt 9.8	1.0 to 2.0
	Yaw	Rt 16.8	Lt 13.2	0.5 to 1.2
Armed	Pitch	Up 4.2	Dn 6.5	1.4
	Roll	Rt 9.8	Lt 11.8	1.2
	Yaw	Rt 10.8	Lt 11.8	0.8

NOTE: Comparison made at 95 KCAS and 9300 pounds gross weight.

2.4 VIBRATION

Qualitatively, the vibration characteristics of the helicopter were generally satisfactory. In calm air the aircraft exhibited a self-induced lateral 2/3-per-rev undamped vibration of varying amplitudes. This vibration, of undetermined origin, resulted in a grounding of the helicopter and termination of testing. The same vibration has been reported on other UH-1B/540 helicopters. This vibration was extremely disconcerting to the crew. The effect this vibration has on component life is not known and, therefore, the vibration could cause a safety-of-flight condition.

2.5 AIRSPEED CALIBRATION

A calibrated trailing bomb was used to determine the position error of the standard airspeed system. The results of this test are shown in figure 68, appendix I. The standard airspeed system met the requirements of MIL-I-6115A at speeds above 40 KCAS.

2.6 JETTISON

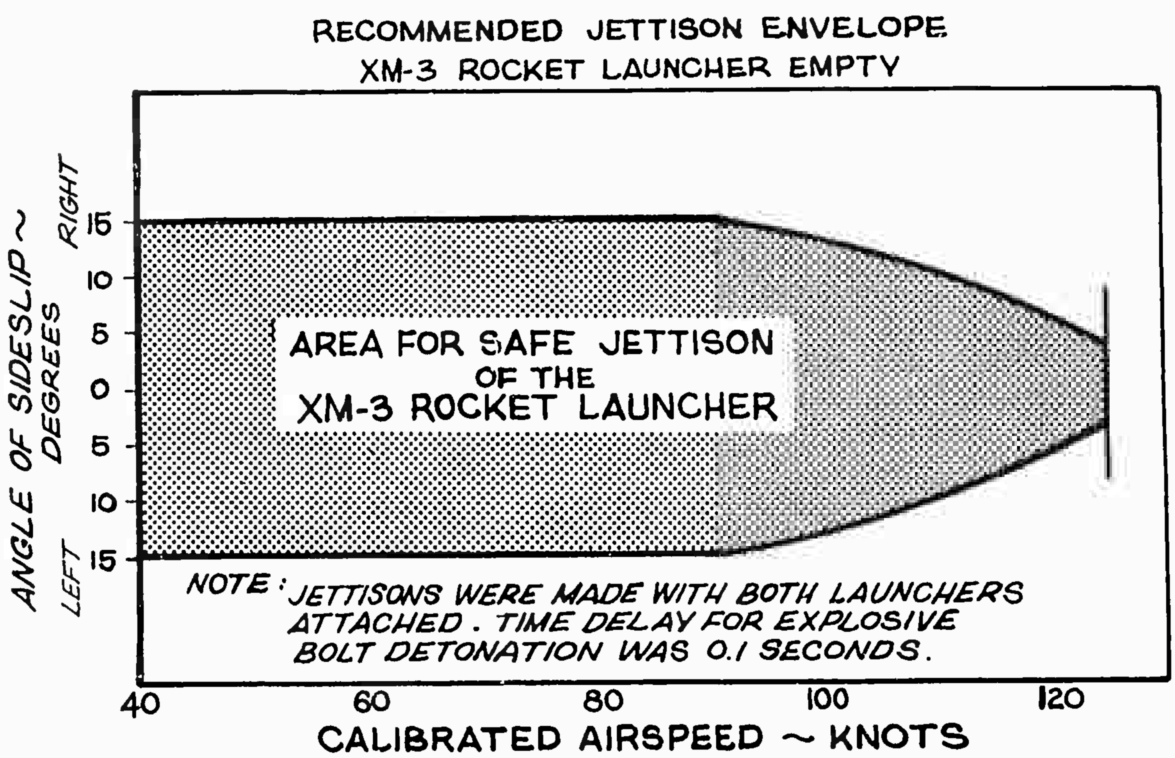
Jettisoning of the XM-3 (2.75-inch) rocket launcher was accomplished to obtain a flight envelope for safe jettison of the launcher. Empty launchers were used since previous tests with loaded and empty launchers indicated that jettison of the empty launchers was the more critical (reference x).

Jettisons were accomplished by detonating two explosive bolts with a 0.1-second time delay between jettison of the top and bottom bolts. These bolts connected the launcher to a crank arm on the universal pylon. Using this time delay allowed the launcher to rotate away from the aircraft before the bottom bolt detonated. Tests were conducted during stabilized flight at a gross weight of 8100 pounds, forward C.G. (126.7 inches), rotor speed of 324 rpm, and an altitude of 500 feet above the ground. The launchers were installed at the maximum elevation angle of 6 degrees noseup from the waterline of the helicopter.

Level flight jettison data indicated the critical flight conditions at which to jettison were from the high side (left side in right sideslip and right side in left sideslip) due to increasing roll attitude with increasing sideslip. The launcher had a yawing tendency away from the aircraft due to the initial roll of the launcher and a pitchdown tendency due to the pitch attitude of the aircraft. When hits were sustained on the skid the launcher fell clear of the aircraft in approximately 0.7 seconds. When hits were not sustained on the skid the launcher cleared the skids in less than 0.5 seconds. At no time did the launcher have a tendency to float or to move aft into the tail section. The general flight characteristics of the launcher, figures 69 and 70, appendix I, are representative of all level flight jettisoning.

The recommended safe jettison envelope during level flight and high-powered descents is shown in figure G. No jettisoning in close proximity to the ground is recommended due to possibility of the aircraft's being damaged from a rebounding launcher.

Figure G LEVEL FLIGHT and HIGH POWER DESCENT



Jettisons during full autorotation were conducted at 60, 70, and 80 KCAS. During these tests the launcher floated on the skid and aft crosstube approximately 0.4 seconds, then left the aircraft as shown in figure 71, appendix I. Due to the nonavailability of sufficient XM-3 rocket launchers a complete autorotation jettison envelope could not be established. If jettisoning is necessary during an autorotation it is recommended that 60 KCAS, zero-degree sideslip be maintained.

2.7 WEAPONS FIRING

Firing tests were conducted to determine if any safety-of-flight or stability and control problems existed. During the tests the M-5 subsystem was a permanent installation on the helicopter; whereas the XM-16, XM-21 or XM-3 subsystem was interchangeable depending upon the test being conducted. The systems were fired at conditions that would produce the maximum adverse gross weight of 9000 pounds, a forward C.G. of 127.4 inches, and a rotor speed of 324 rpm. The speed ranged from zero knots (IGE hover) to 125 KCAS (V_{NE}).

2.7.1 M-5 (M-75 Grenade Launcher)

The M-5 subsystem was fired in the stowed position (zero-degree elevation and zero-degree azimuth) and in combination of 15-degree elevation and ± 60-degree azimuth or 35-degree depression and ± 60-degree azimuth.

When firing the M-5 subsystem no adverse stability and control problems were noted. In level flight the helicopter would pitch or yaw depending on the gun position at speeds below 80 KCAS. This was easily controlled by the pilot. Above 80 KCAS the recoil "shook" the helicopter but no yawing or pitching tendency was evident. Firing during hover at 35-degree depression caused the aircraft to pitch up and translate rearward. Timing marks on the oscillograph trace showed the M-5 firing rate to be 200 rounds/minute. During the firing of the M-5, extreme deflection of the combination sight would cause the weapon to cease firing. The combination sight and/or the M-5 subsystem should be modified to allow the system to fire at maximum deflection.

2.7.2 XM-16 (M-60C Machine Gun)

The XM-16 subsystem was fired in the stowed position (zero-degree elevation and zero-degree azimuth) and in combinations of 9-degree elevation and ±70-degree azimuth or 66-degree depression and ±70-degree azimuth. The system was also fired at 12-degree elevation and 66-degree depression with zero-degree azimuth. The

M-60C machine gun ceased firing when either weapon traversed to its inboard limit (12 degrees).

The system had only one malfunction; this was caused by improper feeding of the ammunition booster sprocket.

2.7.3 XM-21 (XM-134 Mini-Gun)

The XM-21 subsystem was fired in the stowed position (zero-degree elevation and zero-degree azimuth) and in combination of 12-degree elevation $\pm$70-degree azimuth or 85-degree depression and $\pm$70-degree azimuth. As with the XM-16 subsystem, the XM-134 mini-gun ceased firing when at its inboard limit (12 degrees).

Firing the XM-21 at maximum azimuth conditions caused the helicopter to yaw slightly; however, no control problems were encountered. Two malfunctions were caused by failure of the feeder chute to stretch during extreme right or left deflection. This problem was eliminated by allowing more slack to the feeder chute. Another malfunction was caused by the ammunition booster sprocket's becoming jammed. In the stowed position the firing rate was 2600 rounds/minute for the right gun and 2800 rounds/minute for the left gun. With the weapons at maximum left or right azimuth the firing rate was 4000 rounds/minute (left gun only) and 3700 rounds/minute (right gun only). During the rate-of-fire tests of the XM-21 the right gun failed to cease firing when the left gun was at maximum left azimuth. This was a hazardous situation and all personnel should be made aware of this possible condition.

2.7.4 XM-16/XM-21 (XM-157 Rocket Launcher)

The XM-157 rocket launcher which was used with either the XM-16 or XM-21 subsystem was fired at the conditions listed in table III.

No adverse stability and control problems were encountered when firing the XM-157 subsystem. Firing at a ripple rate of 6 pairs/second caused the helicopter to pitch down; however, the pitch was easily controlled.

TABLE III

XM-16/XM-21 FIRING CONDITIONS

Flight Condition	Sideslip Angle deg	Airspeed KCAS	Number of Rockets Fired
Hover (IGE)	0	0	7 per side
Hover (IGE)	0	0	7 Lt Side
Hover (IGE)	0	0	7 Rt Side
Level Flight	0	100	7 per side
Slight Descent (500 fpm)	0	125	7 per side
Slight Descent (500 fpm)	0	125	7 Rt Side
Slight Descent (500 fpm)	0	123	7 Lt Side
Slight Descent (500 fpm)	15 Right	125	7 Rt Side
Slight Descent (500 fpm)	13 Left	120	7 Lt Side

NOTE: Rocket launcher elevated 6 degrees from waterline.

2.7.5 XM-3 Rocket Launcher

The XM-3 rocket launcher was fired at the conditions listed in table IV. The number of rockets fired does not include those fired during buildup to critical conditions. The rocket launcher was installed at the maximum elevation of 6 degrees during stability and control tests and at the maximum depression of 6 degrees during tests to determine the effect of rocket motor debris striking the helicopter.

Salvoing 48 rockets at the rate of 6 pairs/second at 125 KCAS (V_{NE}) caused the helicopter to pitch down slightly. This pitchdown was easily controlled. Salvoing 48 rockets from a hover required approximately 1.1 inches of aft longitudinal control to maintain a constant attitude (figure 72, appendix I). Firing 24 rockets asymmetrically from a hover with 24 dummy rockets installed in the opposite launcher required approximately 1/8 to 1/4 inch of directional and lateral control movement to maintain a constant attitude. The pitchdown and yaw were not considered objectionable. The XM-3 subsystem when deflected 6 degrees down should not be fired during

TABLE IV

XM-3 FIRING CONDITIONS

Flight Condition	Sideslip Angle deg	Airspeed KCAS	Number of Rockets Fired
Hover (IGE)	0	0	24 per side
Hover (IGE)	0	0	24 Lt Side
Hover (IGE)	0	0	24 Rt Side
Slight Descent (500 fpm)	0	125	24 per side
Slight Descent (500 fpm)	5 Left	125	24 Lt Side
Minimum Power Dive (10 deg)	0	125	24 per side
Throttle Chop	0	125	24 per side

NOTE: Rocket launcher elevated 6 degrees from waterline.

IGE hover. If fired IGE the debris created from the rocket detonation upon impact with the ground could damage the aircraft.

During the XM-3 firing tests, the rocket motor debris caused random superficial scratches on the elevator, fuselage, vertical tail, and tail rotor. An improved elevator which had a protective steel shield on the leading edge and an increased strength main spar was installed on the test aircraft. Tests were conducted at different flight conditions than those which caused severe damage to the elevator on other UH-1 series aircraft; therefore, the effectiveness of the steel shield against rocket debris damage is not known.

SECTION 3. APPENDICES

APPENDIX I TEST DATA

FIGURE NO. 1
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

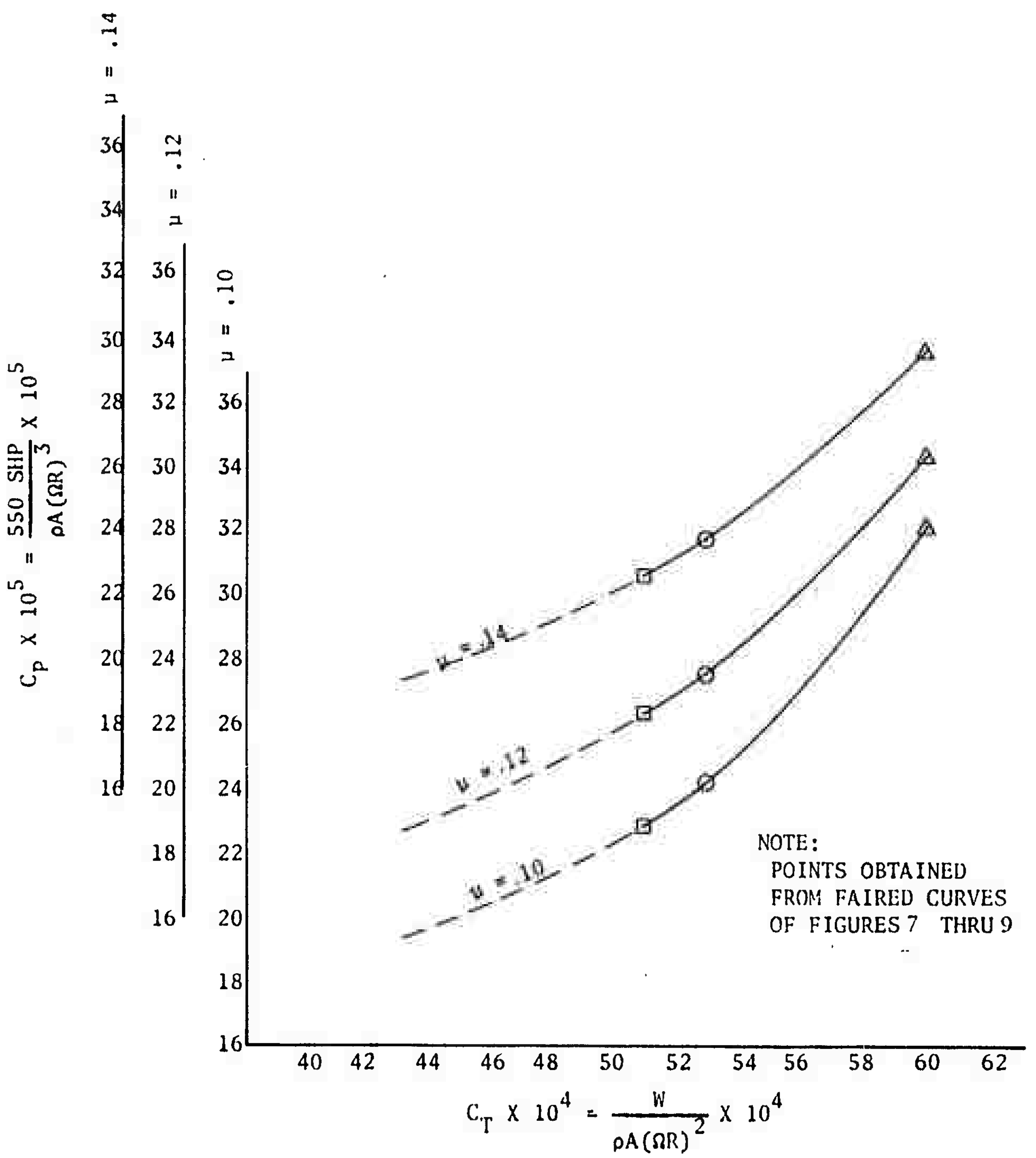


FIGURE NO. 2
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

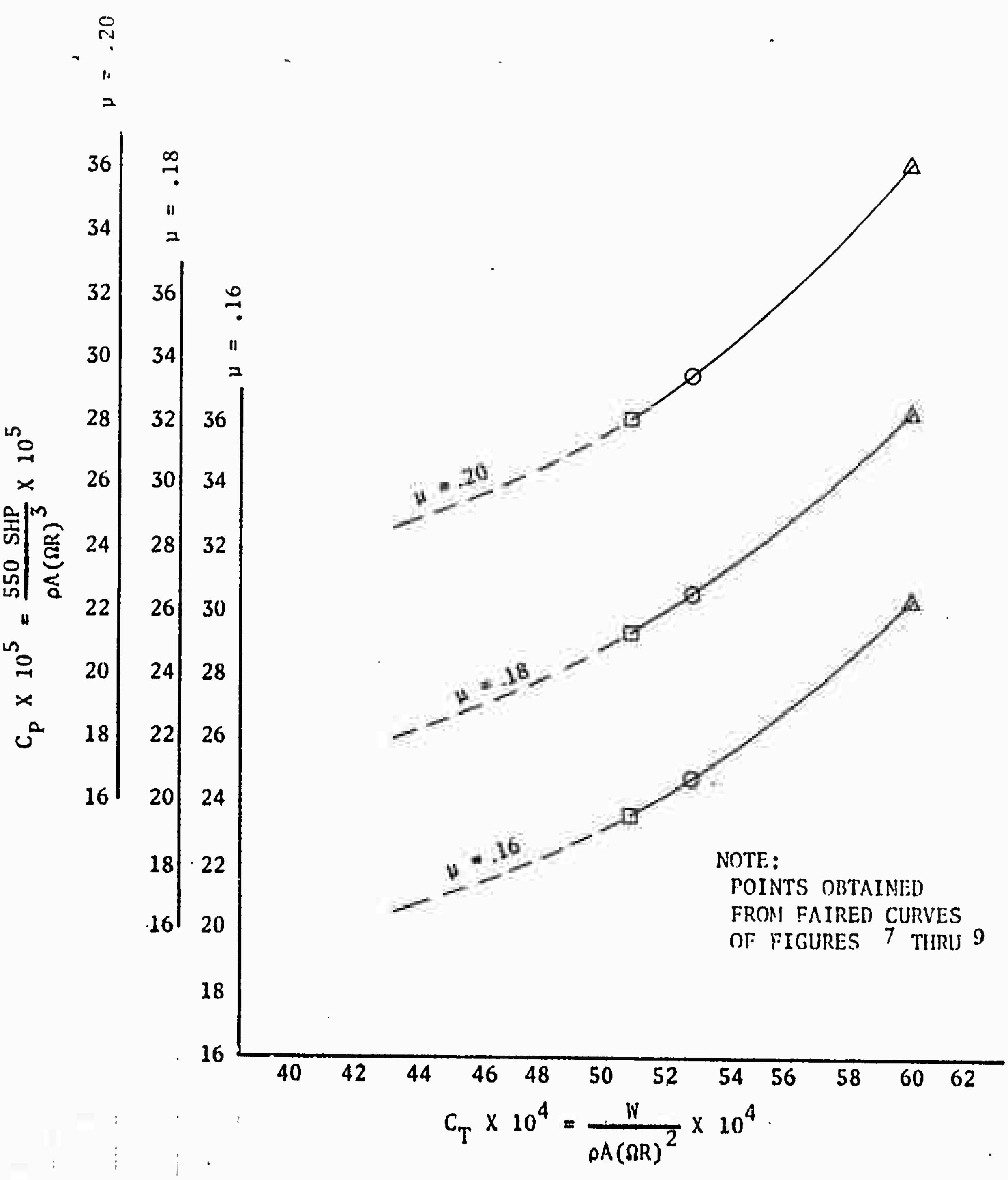


FIGURE NO. 3
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

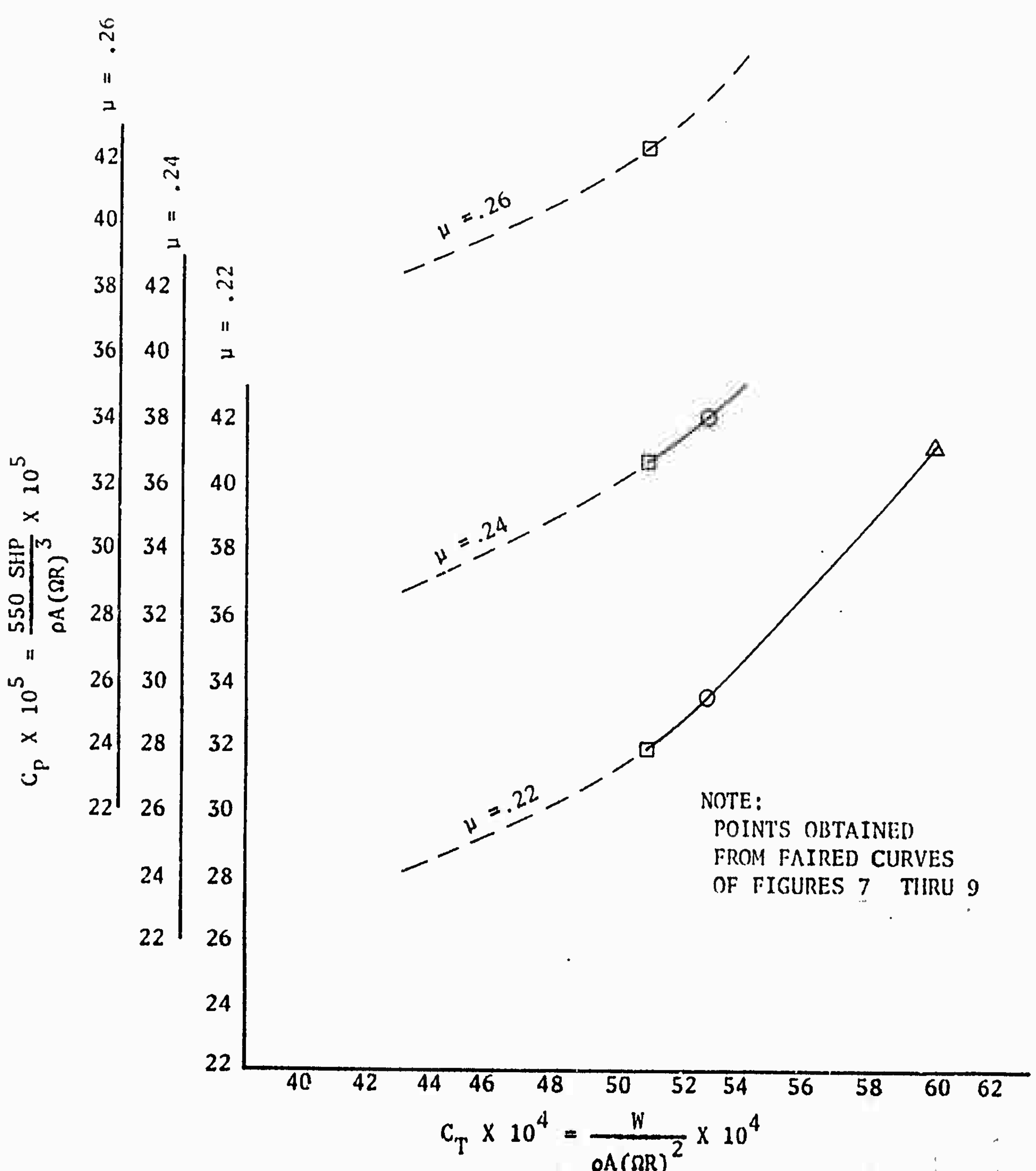


FIGURE NO. 4
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
M-3/M-5 ARMAMENT SUBSYSTEM

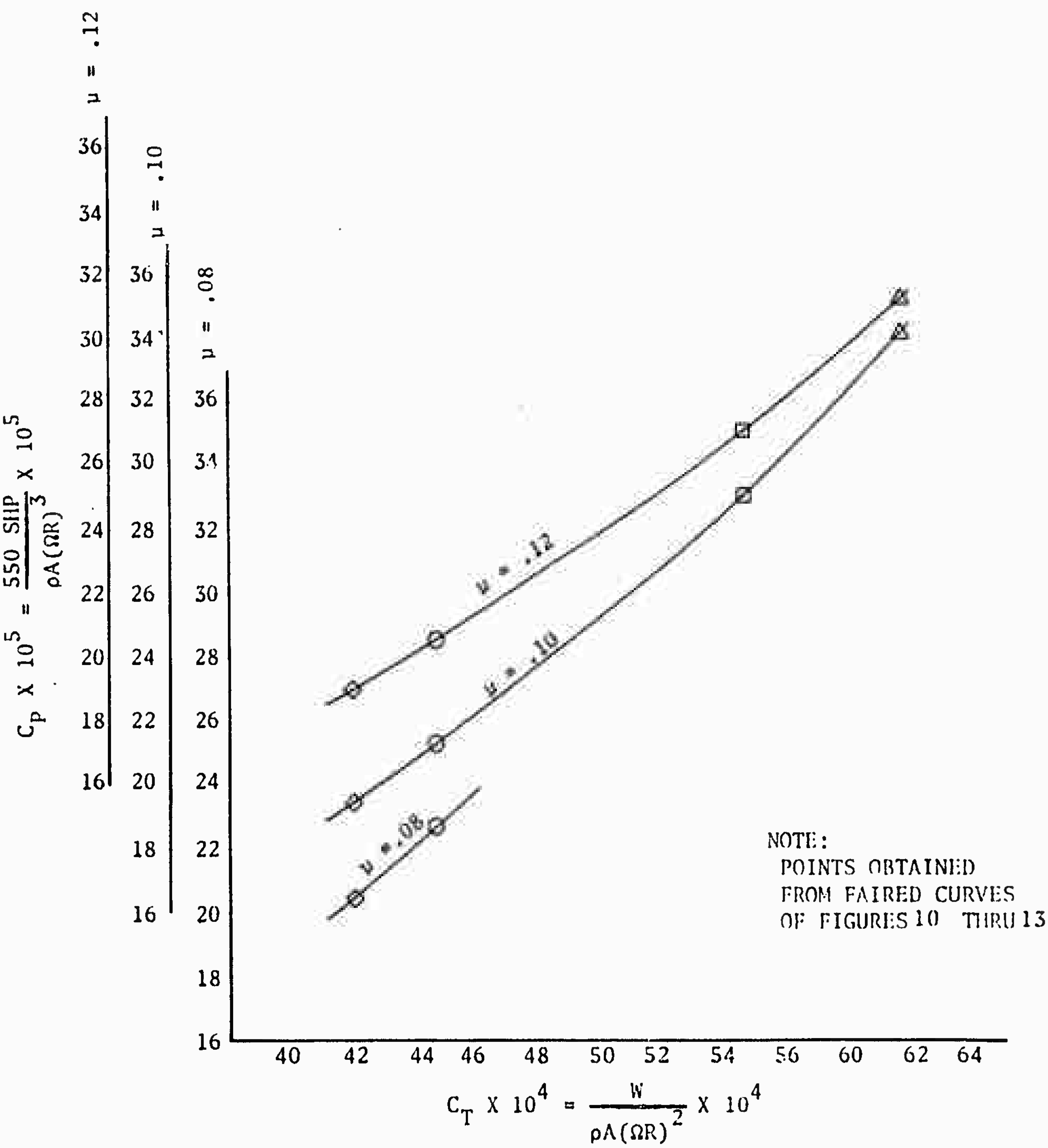


FIGURE NO. 5
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
M-3/M-5 ARMAMENT SUBSYSTEM

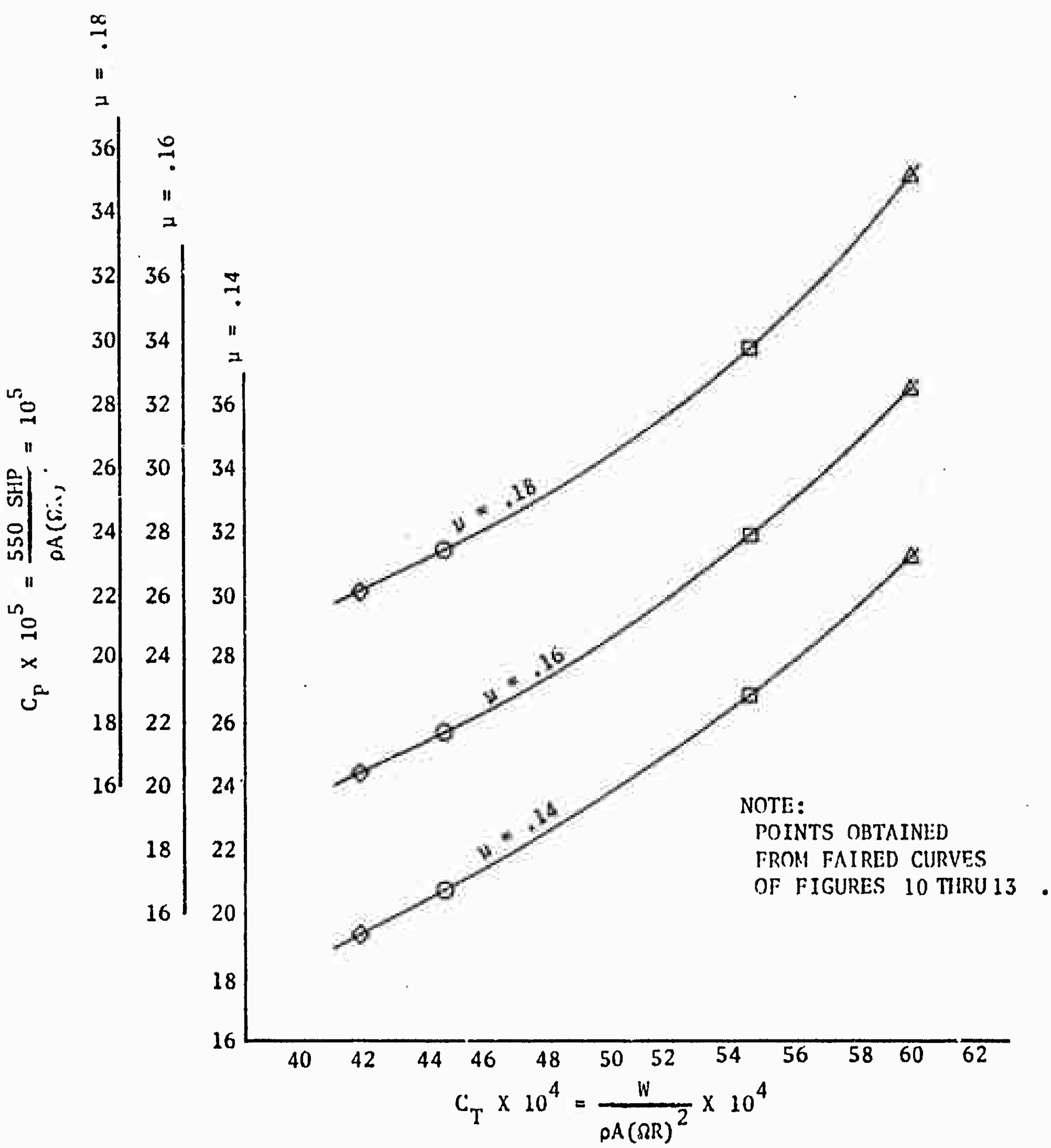


FIGURE NO. 6
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105 +
T53-L-11 ENGINE
M-3/M-5 ARMAMENT SUBSYSTEM

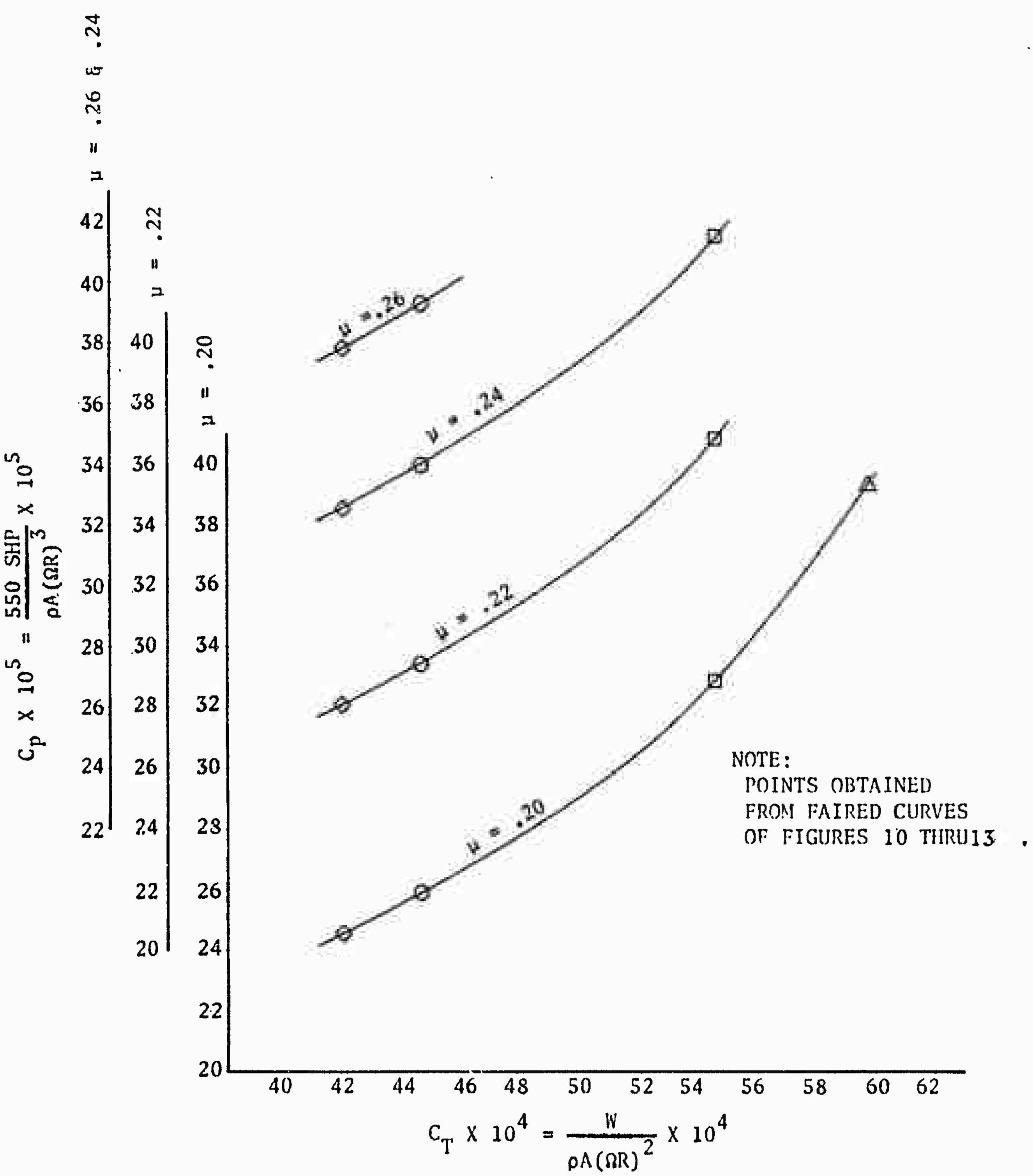


FIGURE NO. 7
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 8990 LBS
DENSITY ALTITUDE = 4360 FT
ROTOR SPEED = 324.0 RPM
C.G. LOCATION = 126.4 IN. (FWD)
C_T = .005078

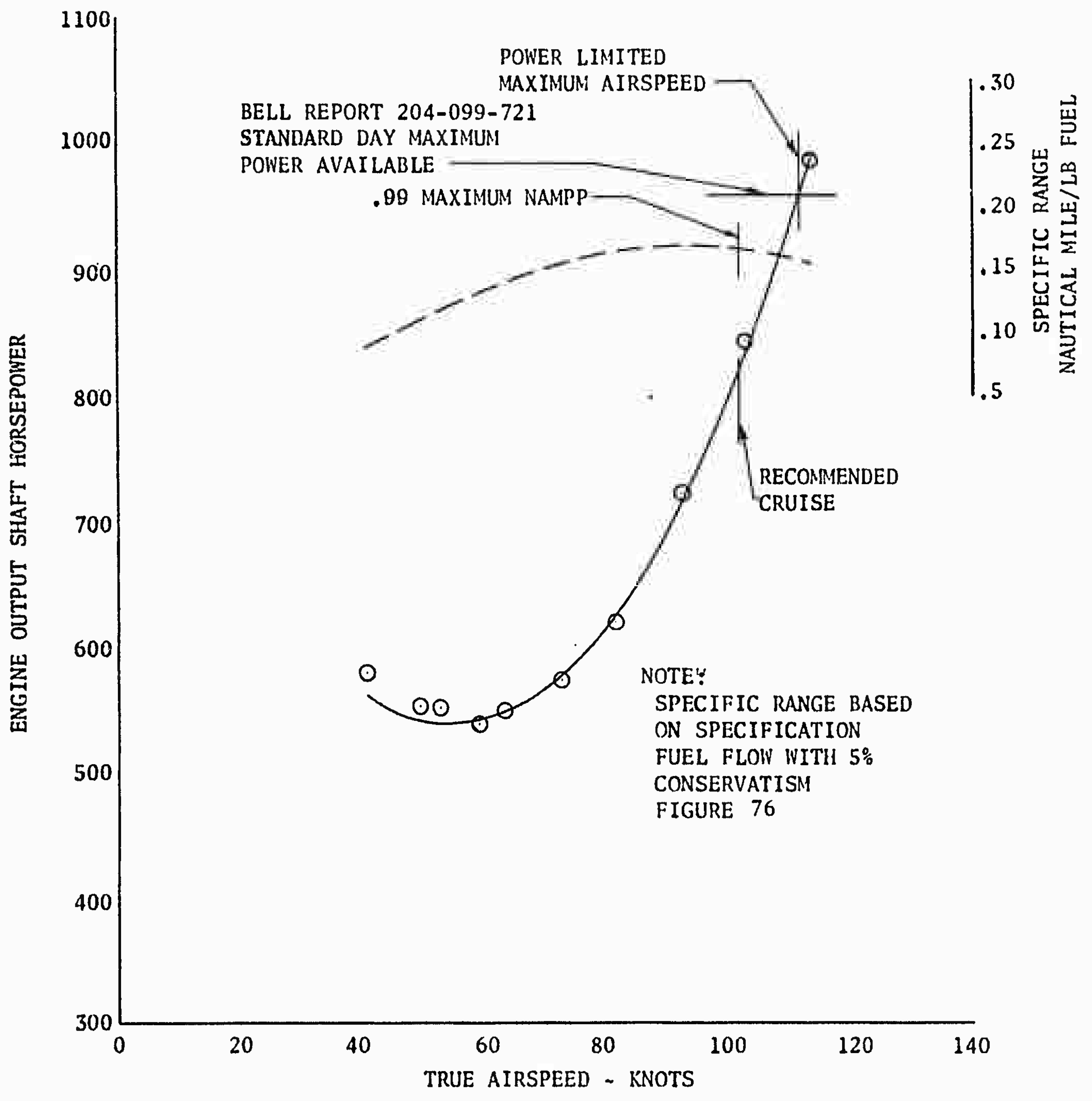


FIGURE NO. 8
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 8090 LBS
DENSITY ALTITUDE = 8995 FT
ROTOR SPEED = 324.0 RPM
C.G. LOCATION = 126.7 IN. (FWD)
C_T = .005273

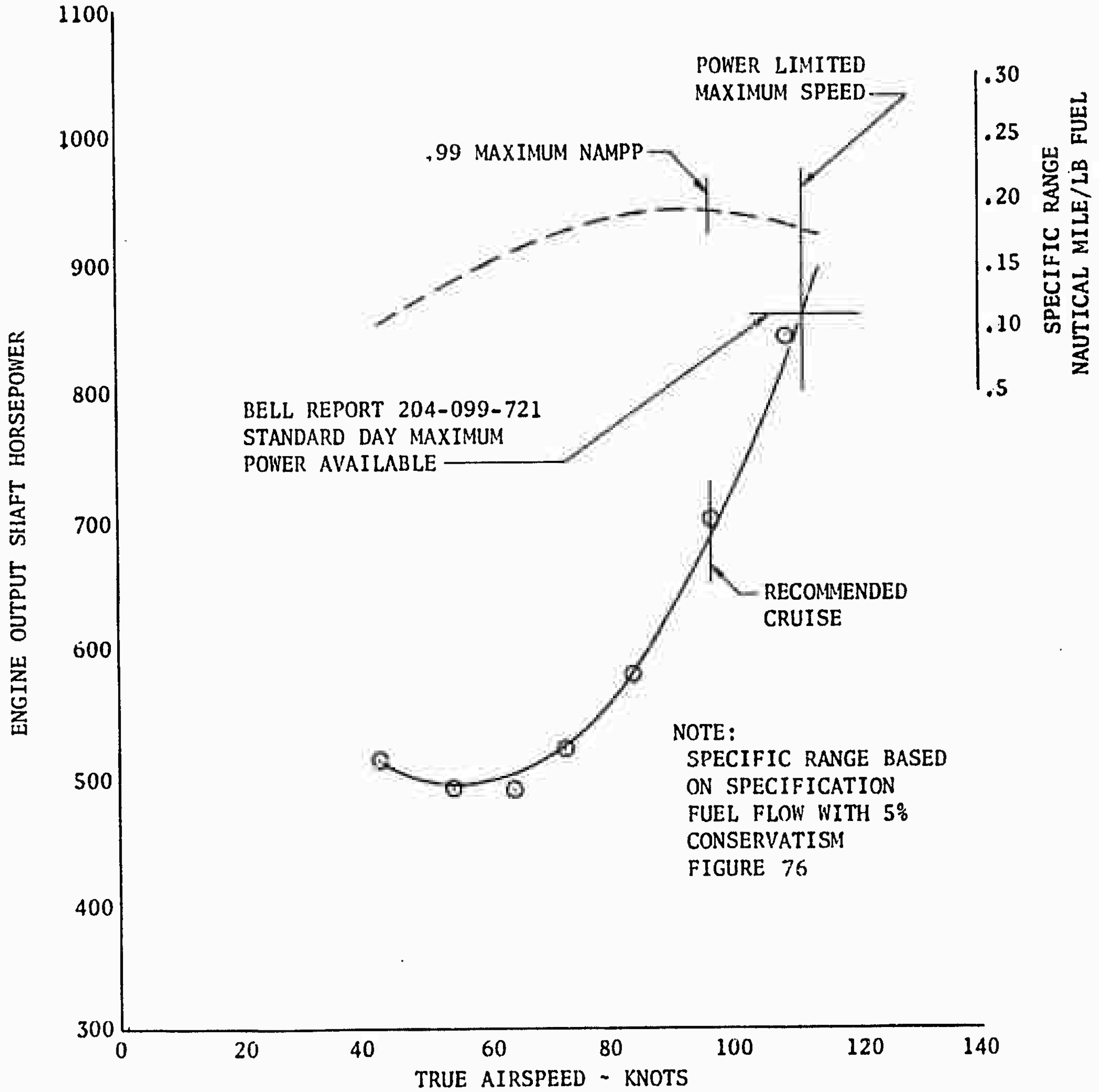


FIGURE NO. 9
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9195 LBS
DENSITY ALTITUDE = 9060 FT
ROTOR SPEED = 324.5 RPM
C.G. LOCATION = 126.4 IN. (FWD)
C_T = .005984

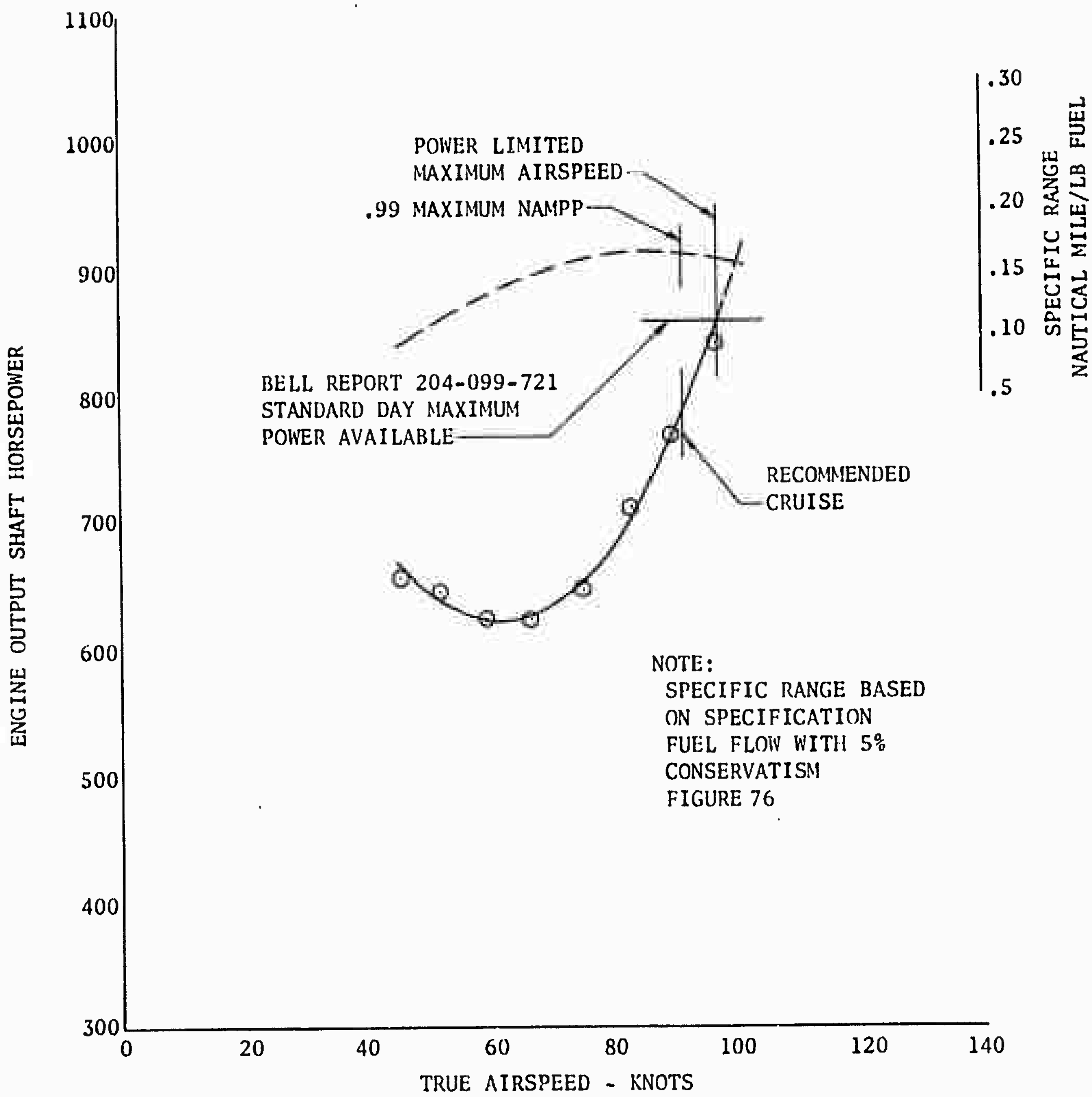


FIGURE NO. 10
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7570 LBS
DENSITY ALTITUDE = 3640 FT
ROTOR SPEED = 324.0 RPM
C.G. LOCATION = 126.3 IN. (FWD)
C_T = .004184

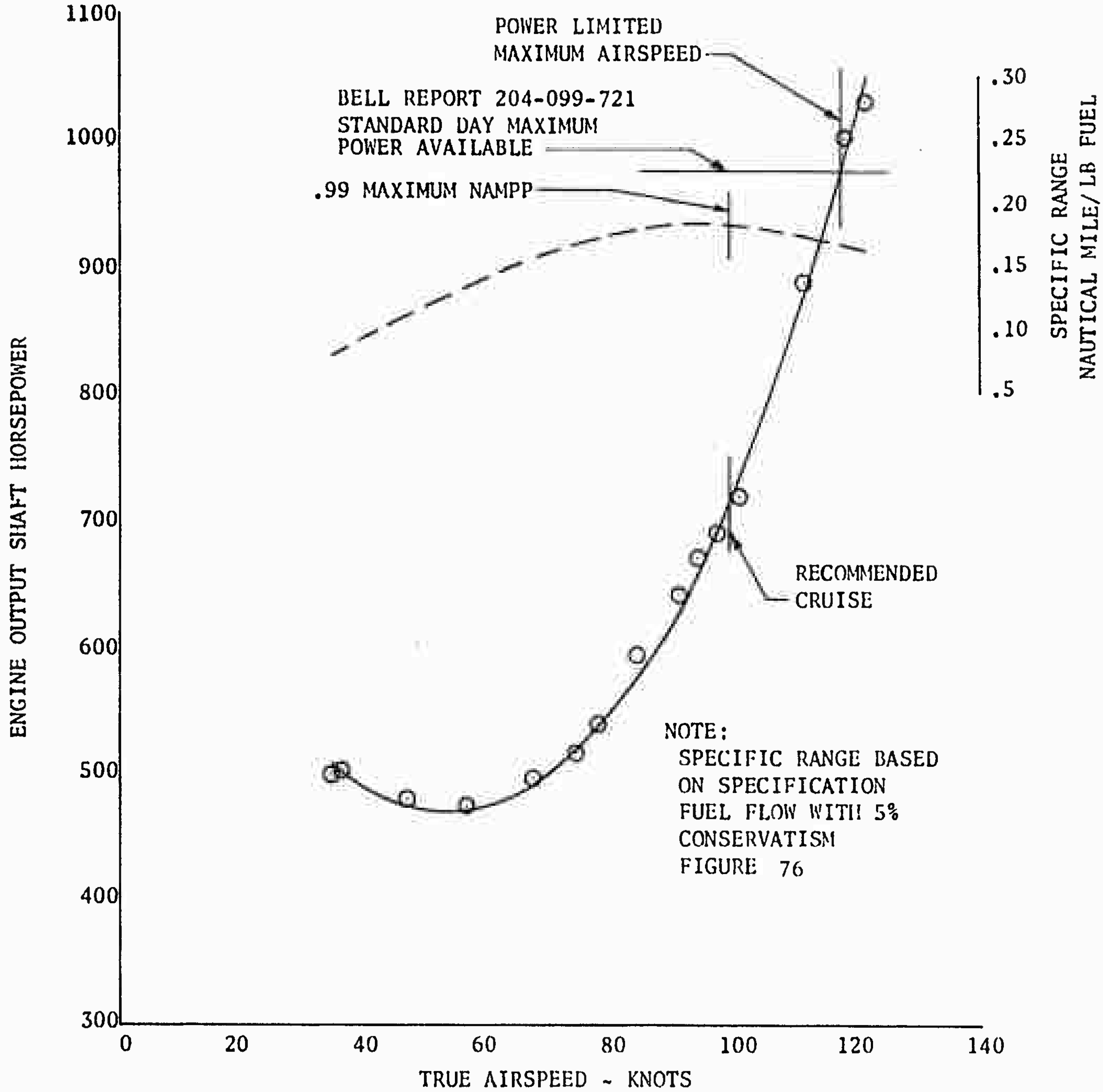


FIGURE NO. 11
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7860 LBS
DENSITY ALTITUDE = 4455 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.5 IN. (FWD)
C_T = .004454

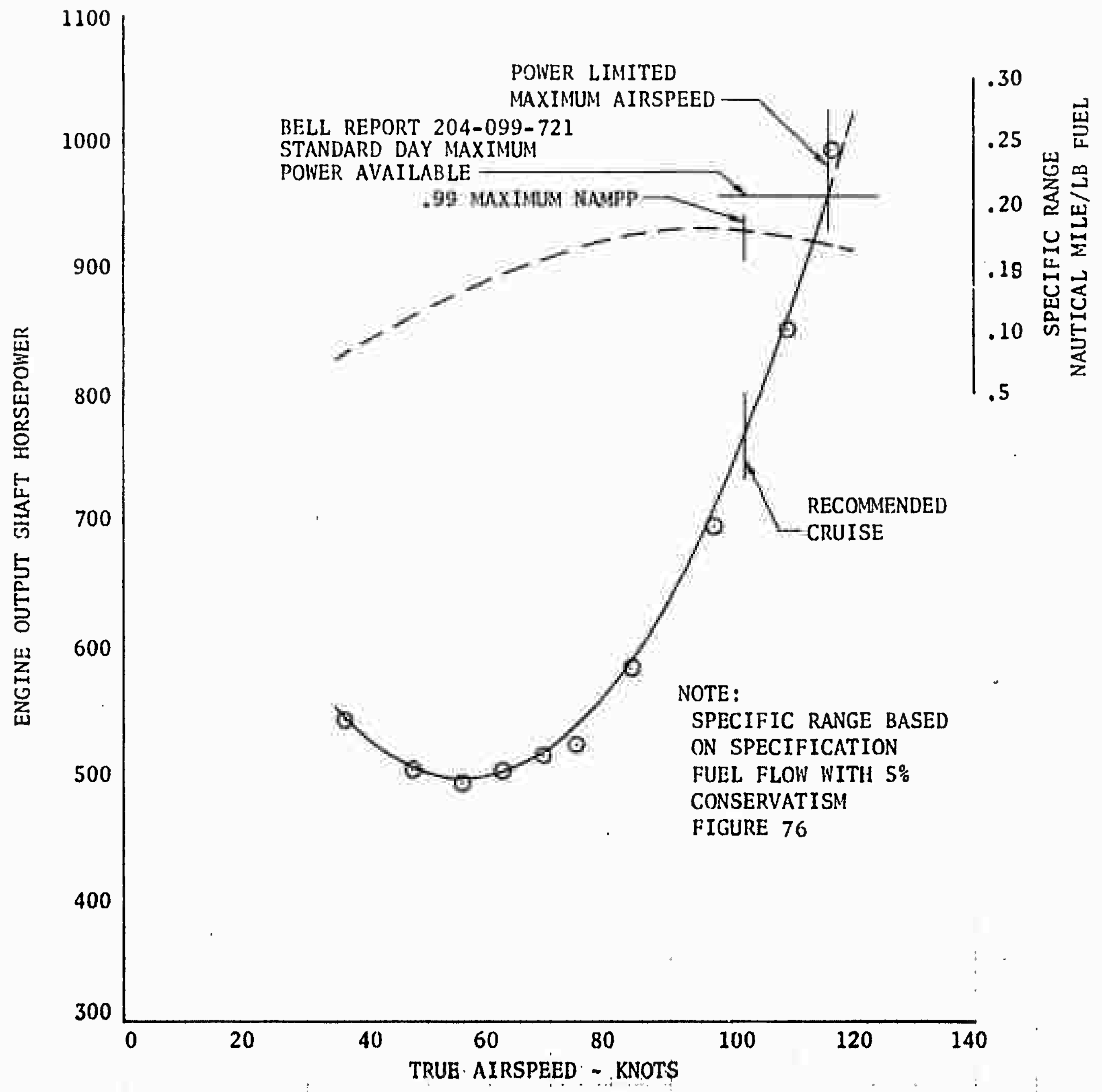


FIGURE NO. 12
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9205 LBS
DENSITY ALTITUDE = 6000 FT
ROTOR SPEED = 324.5 RPM
C.G. LOCATION = 126.2 IN. (FWD)
C_T = .005450

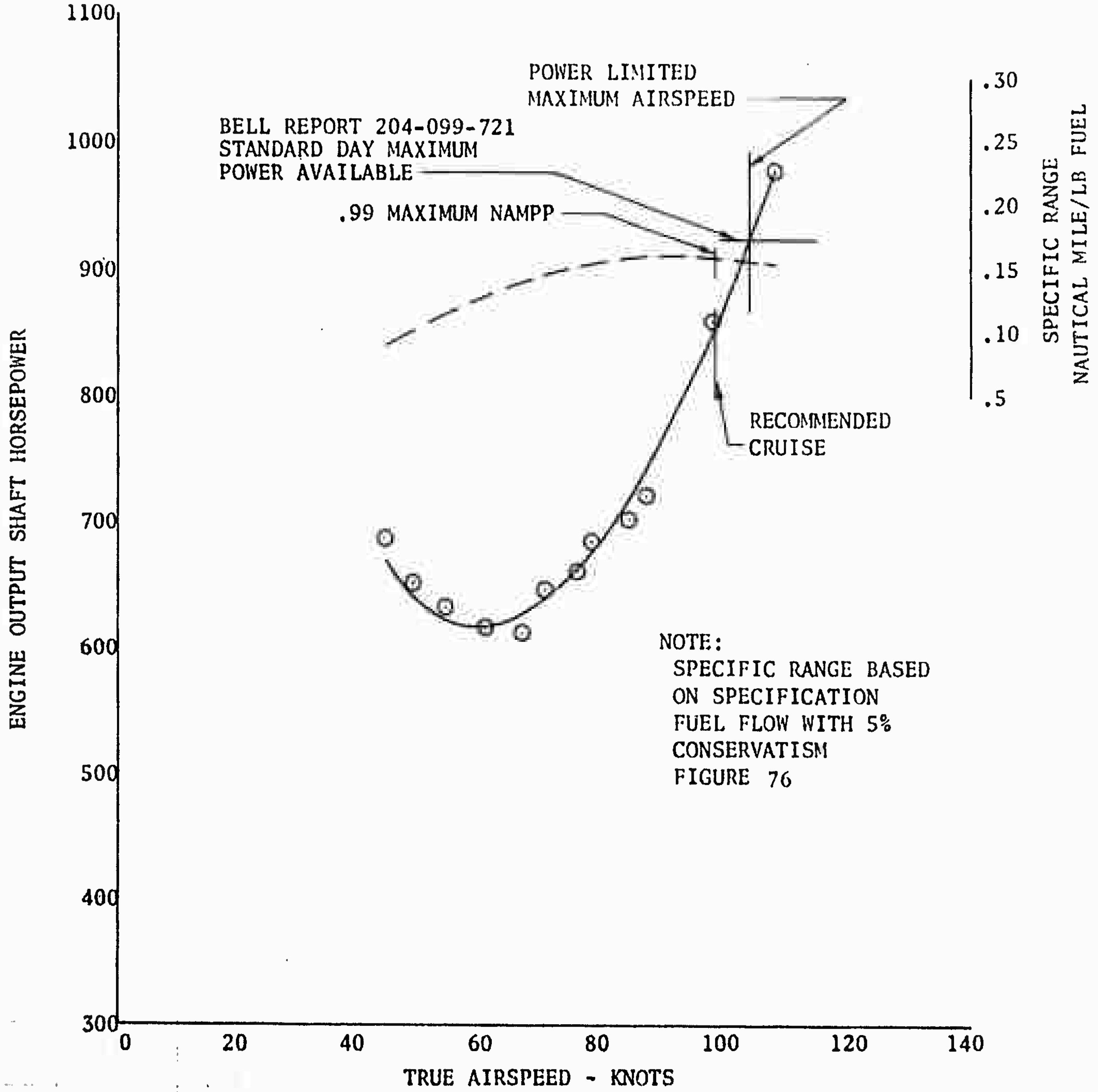


FIGURE NO. 13
LEVEL FLIGHT PERFORMANCE
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9215 LBS
DENSITY ALTITUDE = 8915 FT
ROTOR SPEED = 324.0 RPM
C.G. LOCATION = 126.2 IN. (FWD)
C_T = .005972

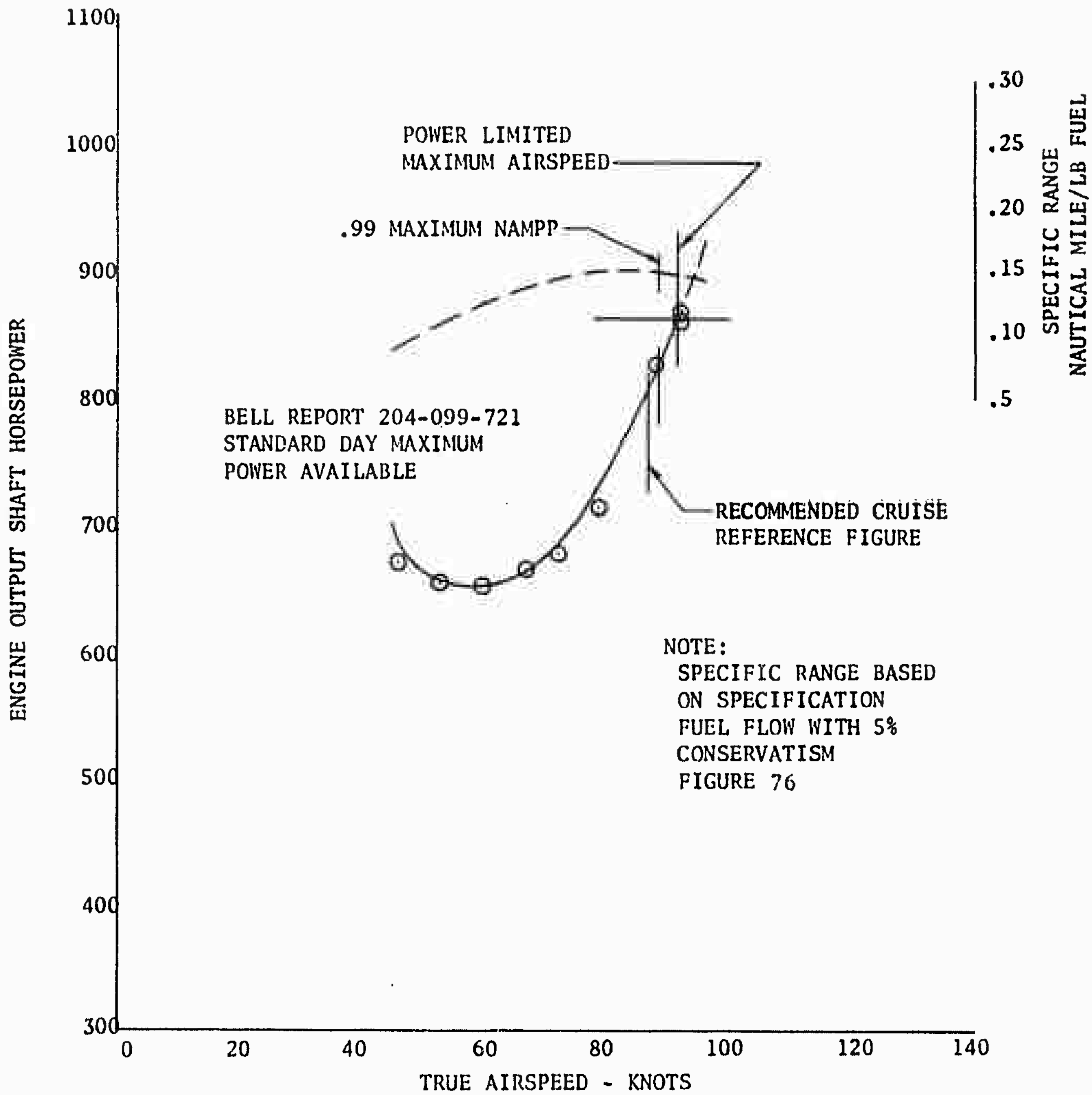


FIGURE NO. 14
LEVEL FLIGHT SUMMARY
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-21/M-5 ARMAMENT SUBSYSTEM
STANDARD DAY DENSITY ALTITUDE - 5000 FT
ROTOR SPEED - 324 RPM

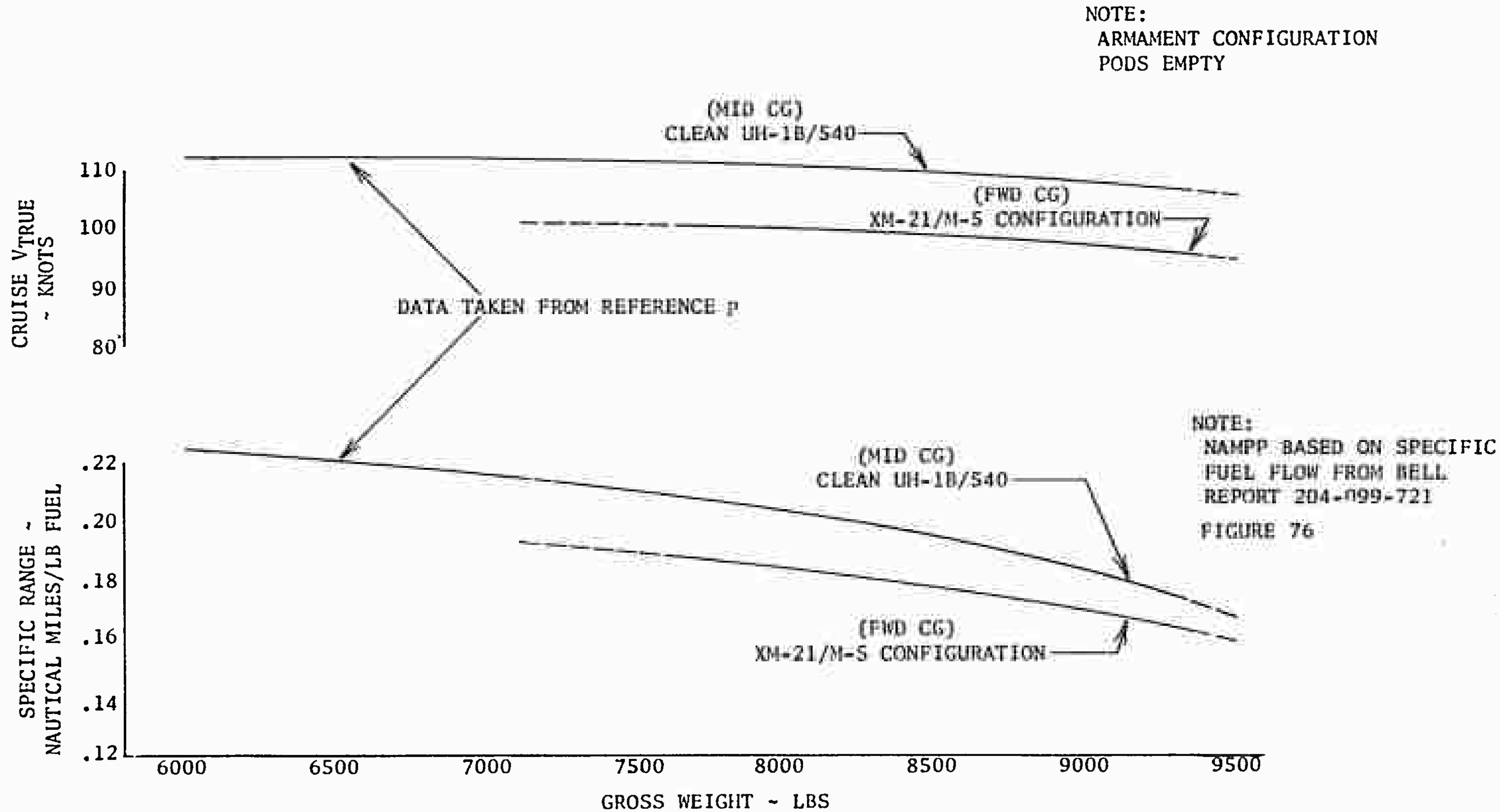


FIGURE NO. 15
LEVEL FLIGHT SUMMARY
UH-1B/540 USA S/N 64-14105
T53-L-11 ENGINE
XM-3/M-5 ARMAMENT SUBSYSTEM
STANDARD DAY DENSITY ALTITUDE - 5000 FT
ROTOR SPEED - 324 RPM

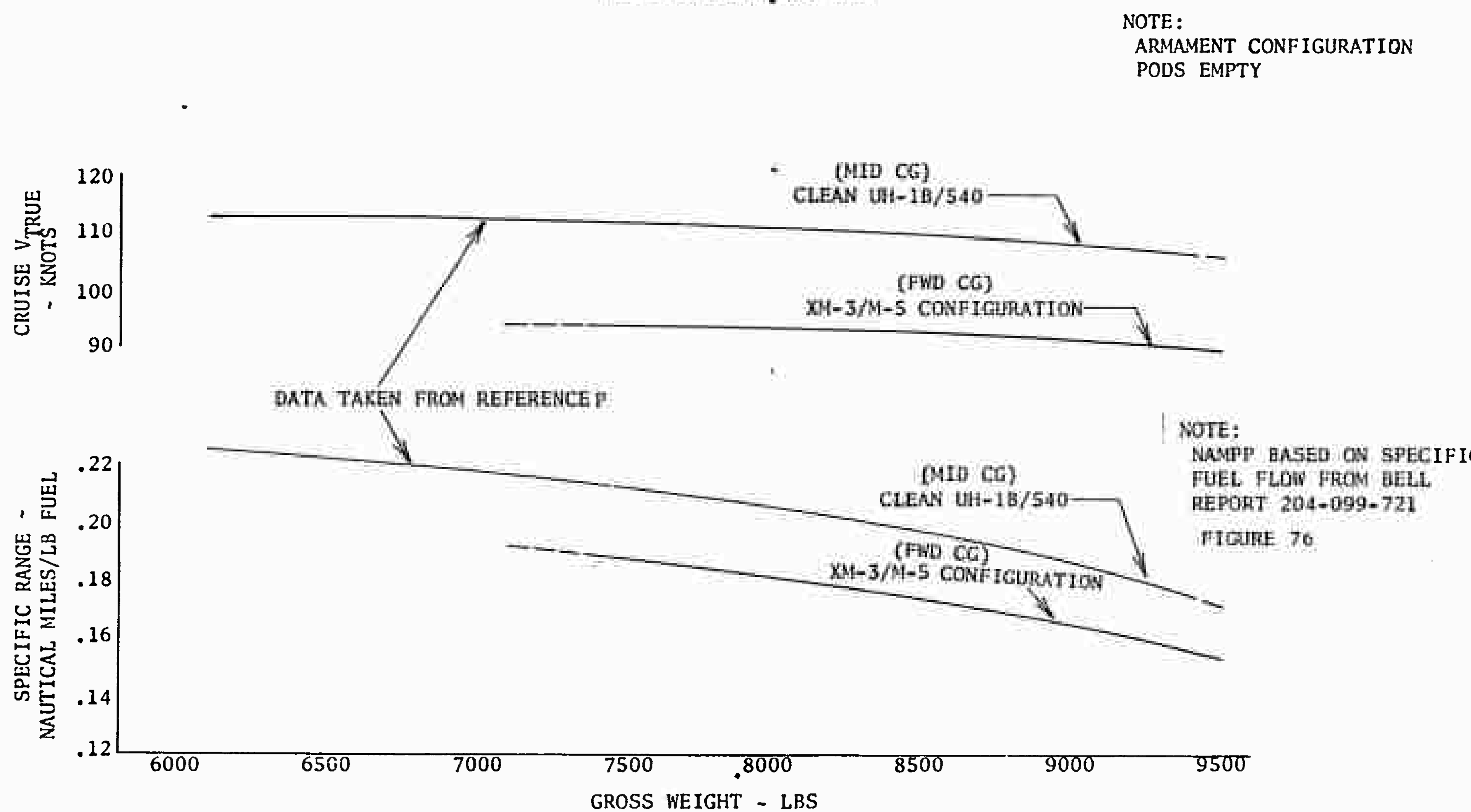


FIGURE NO. 16
AUTOROTATION CHARACTERISTICS
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	ROTOR SPEED RPM	C.G. LOCATION IN	DENSITY ALTITUDE FT	ARMAMENT SUBSYSTEM
□	9330	324	126.4 (FWD)	5000	XM-16/M-5
○	7830	324	126.5	5000	XM-21/M-5
△	7795	324	126.6	5000	M-3/M-5

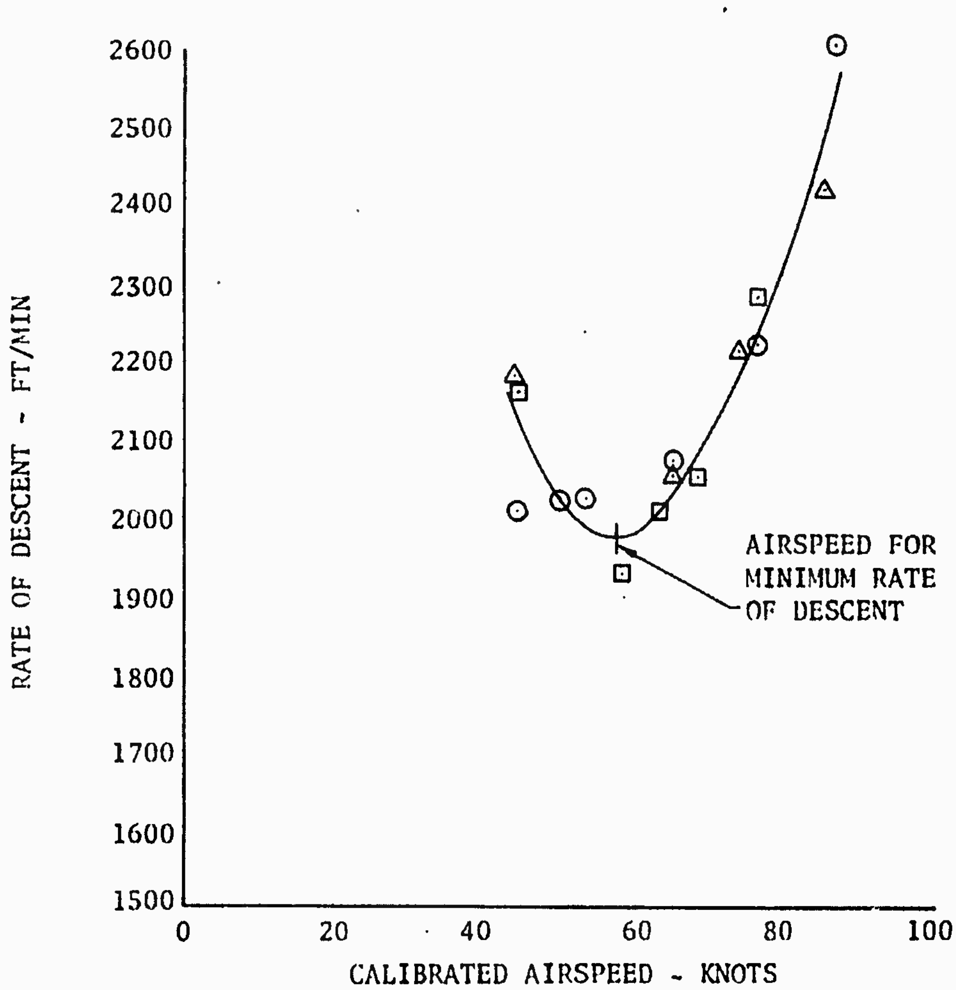


FIGURE NO. 17
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 8090 LBS
DENSITY ALTITUDE = 8995 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.7 IN. (FWD)

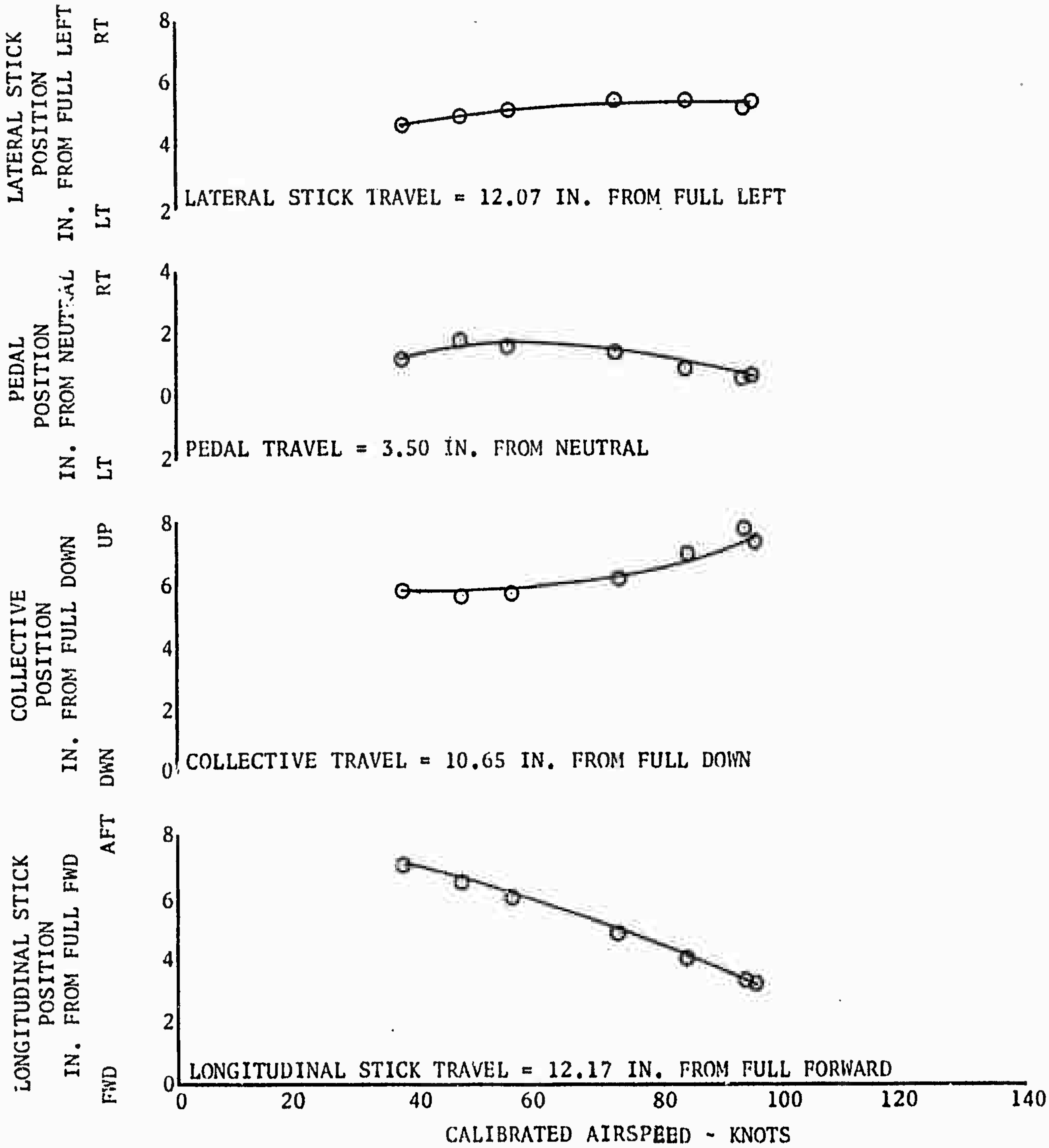


FIGURE NO. 18
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9195 LBS
DENSITY ALTITUDE = 9060 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN. (FWD)

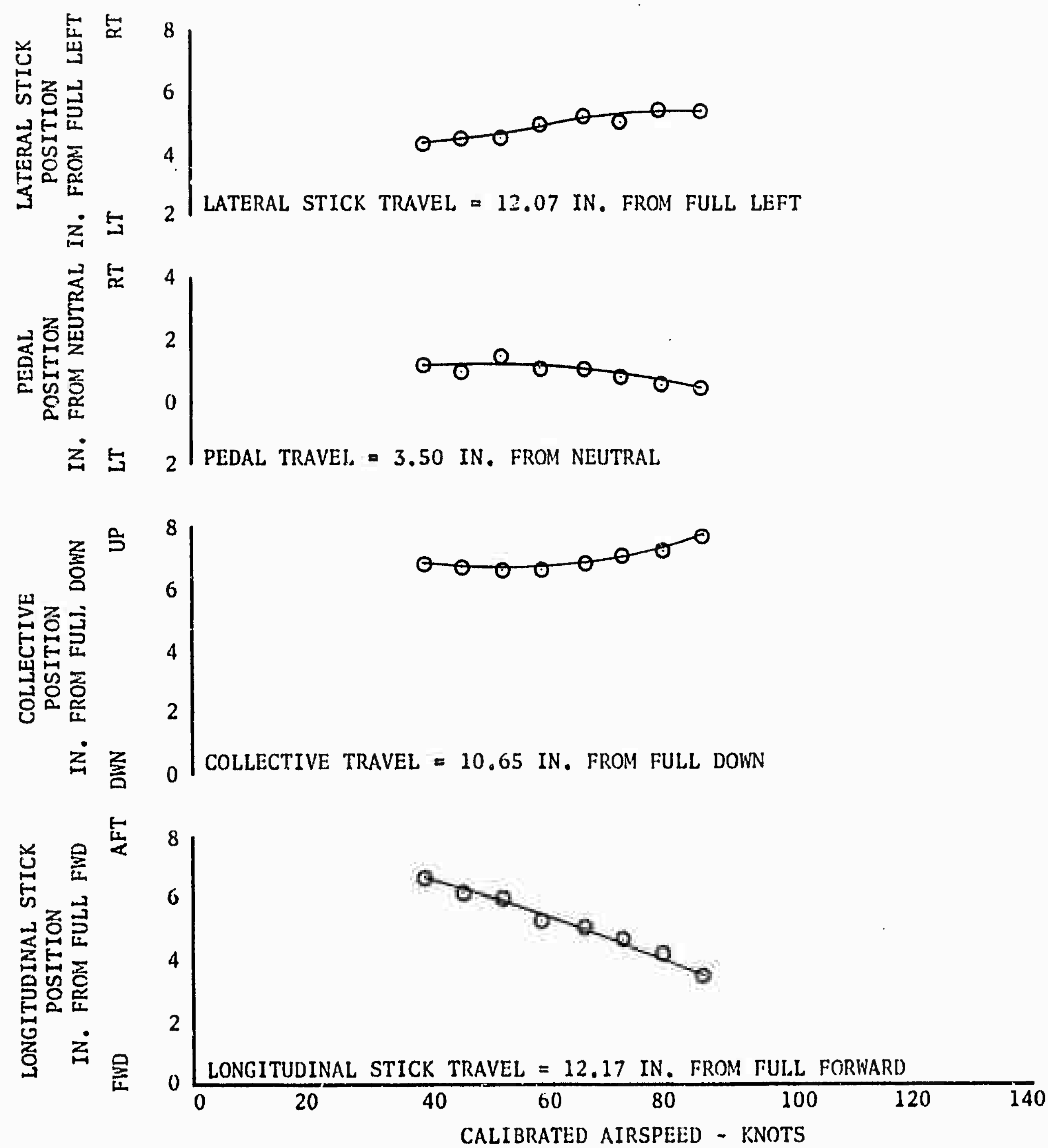


FIGURE NO. 19
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 8990 LBS
DENSTY ALTITUDE = 4360 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN. (FWD)

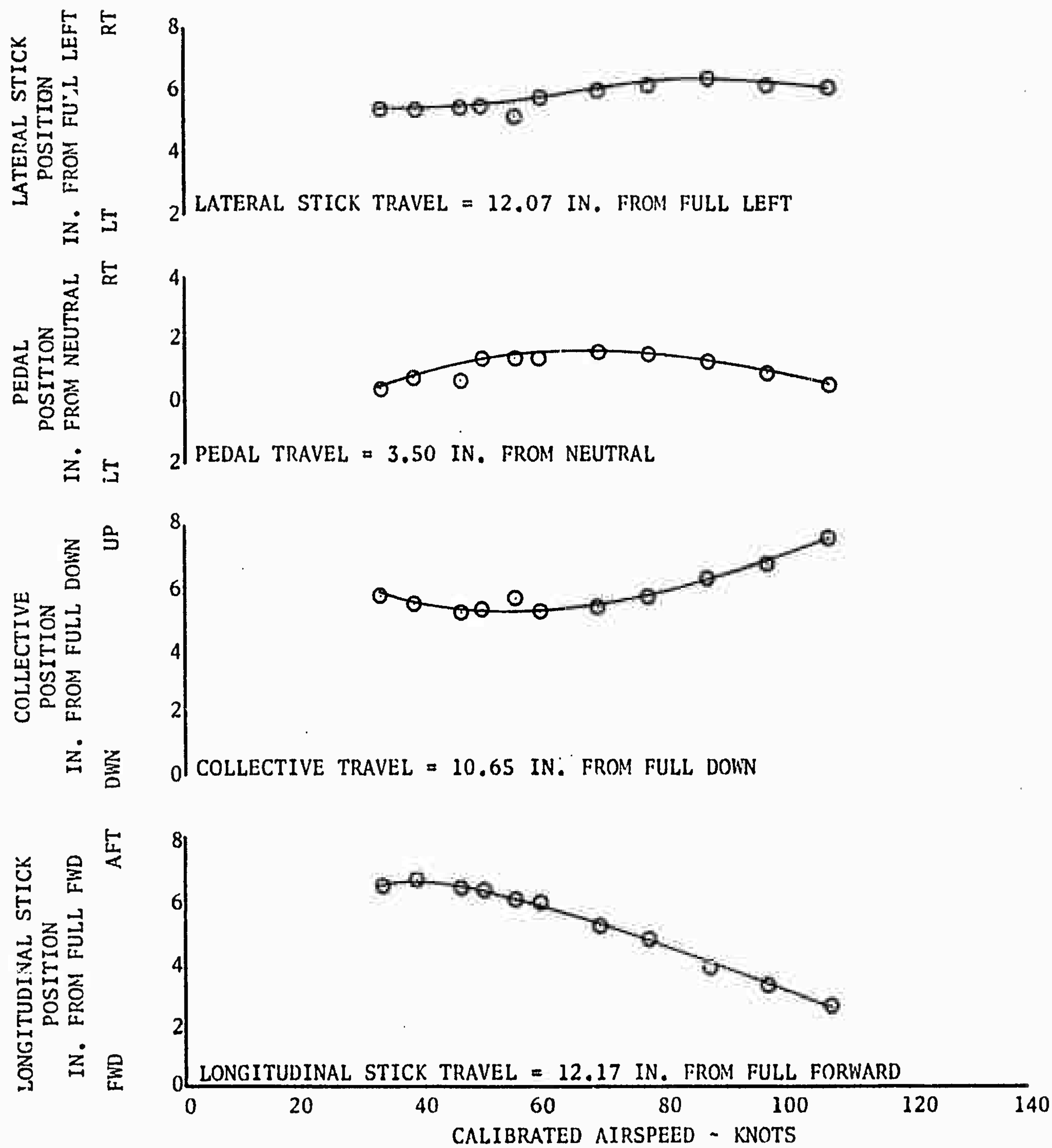


FIGURE NO. 20
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-16/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9290 LBS
DENSITY ALTITUDE = 5720 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN. (FWD)

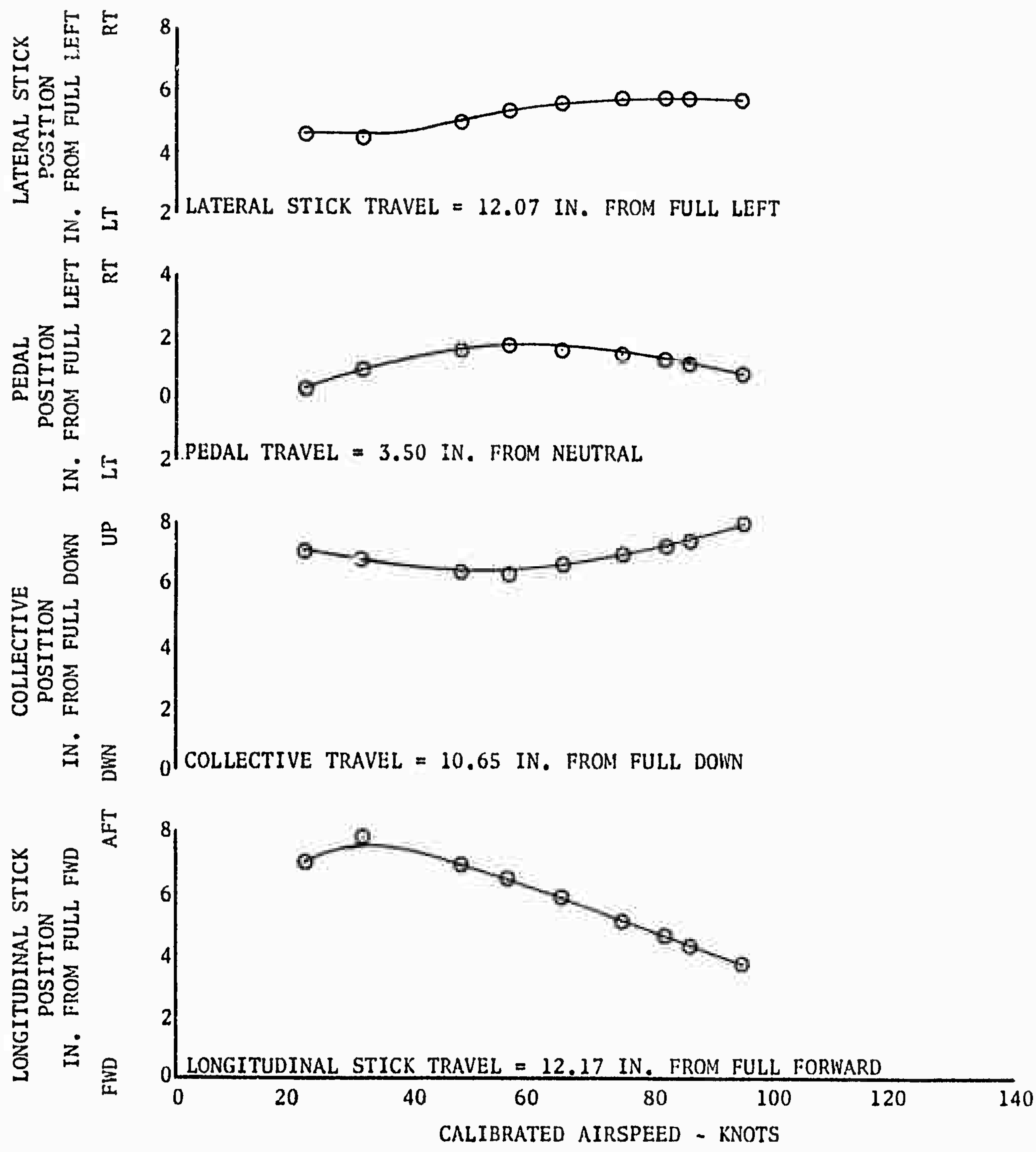


FIGURE NO. 21
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7860 LBS
DENSITY ALTITUDE = 4455 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.5 IN. (FWD)

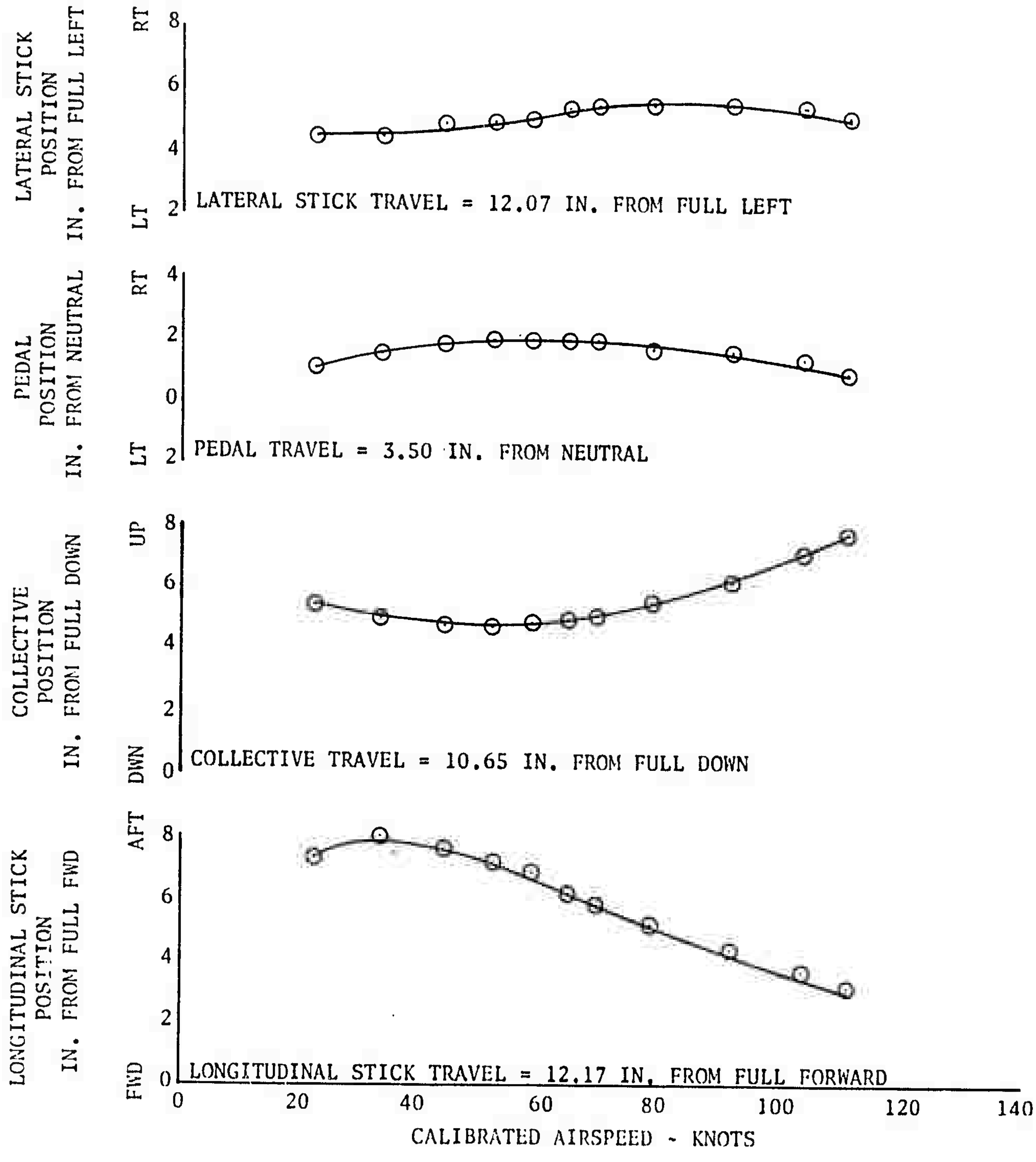


FIGURE NO. 22
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9205 LBS
DENSITY ALTITUDE = 6000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.2 IN. (FWD)

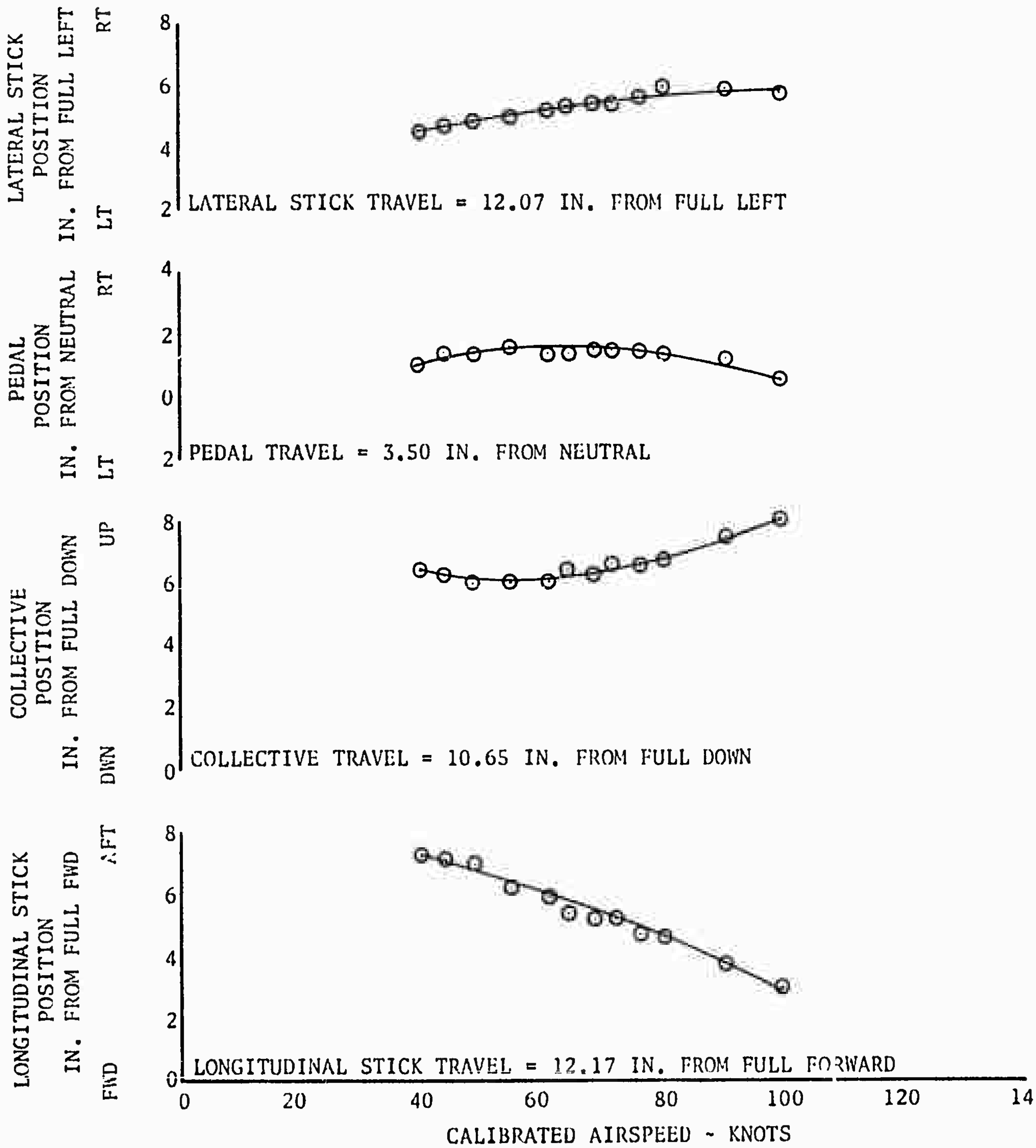


FIGURE NO. 23
CONTROL POSITION TRIM CURVES
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9215 LBS
DENSITY ALTITUDE = 8915 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.2 IN. (FWD)

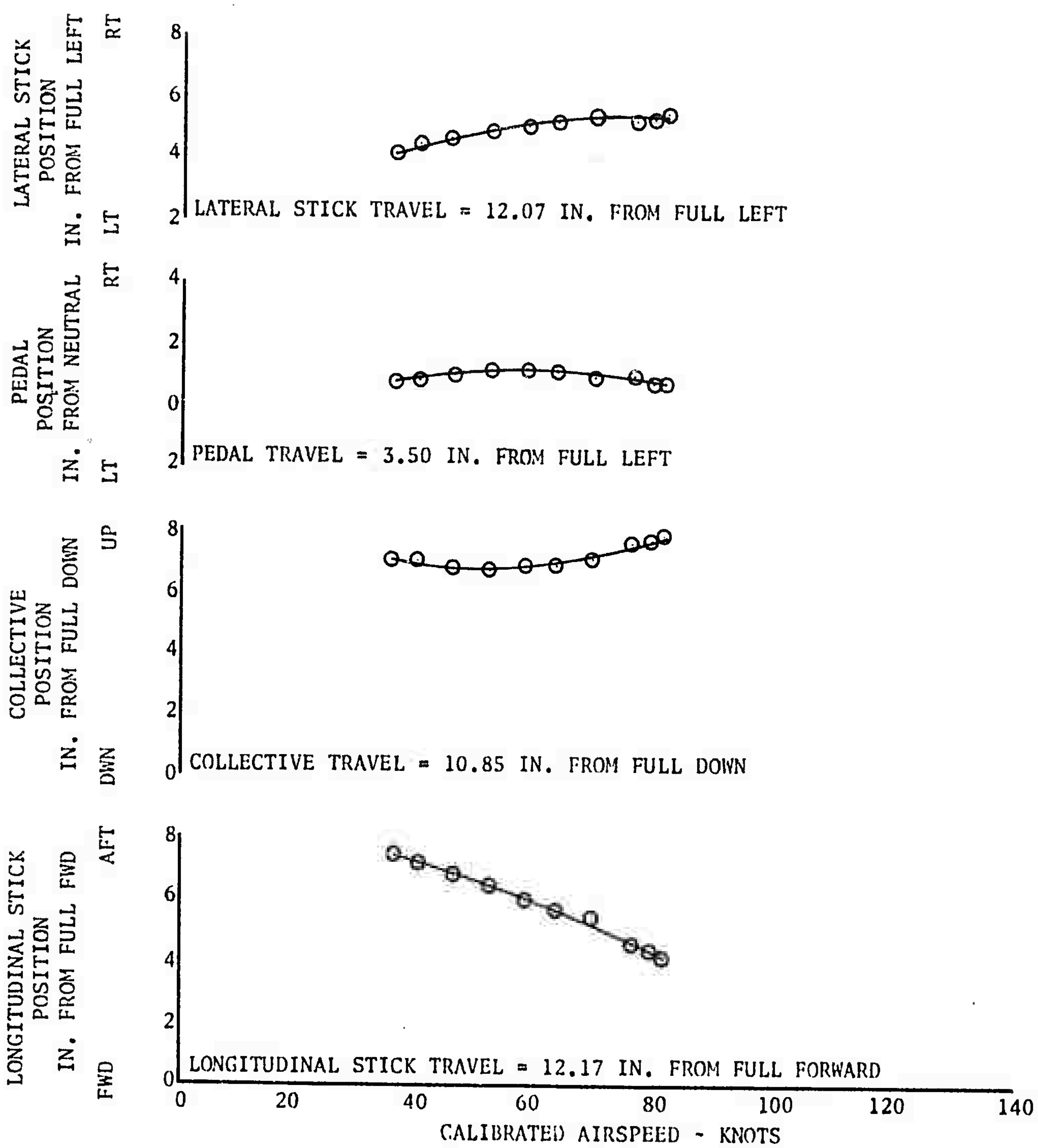


FIGURE NO. 24
STATIC LONGITUDINAL SPEED STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	GROSS WEIGHT LBS
○	39.5	5080	126.5	324	9266
△	81	4950	126.6	324	9335
□	103	4940	126.4	324	9185

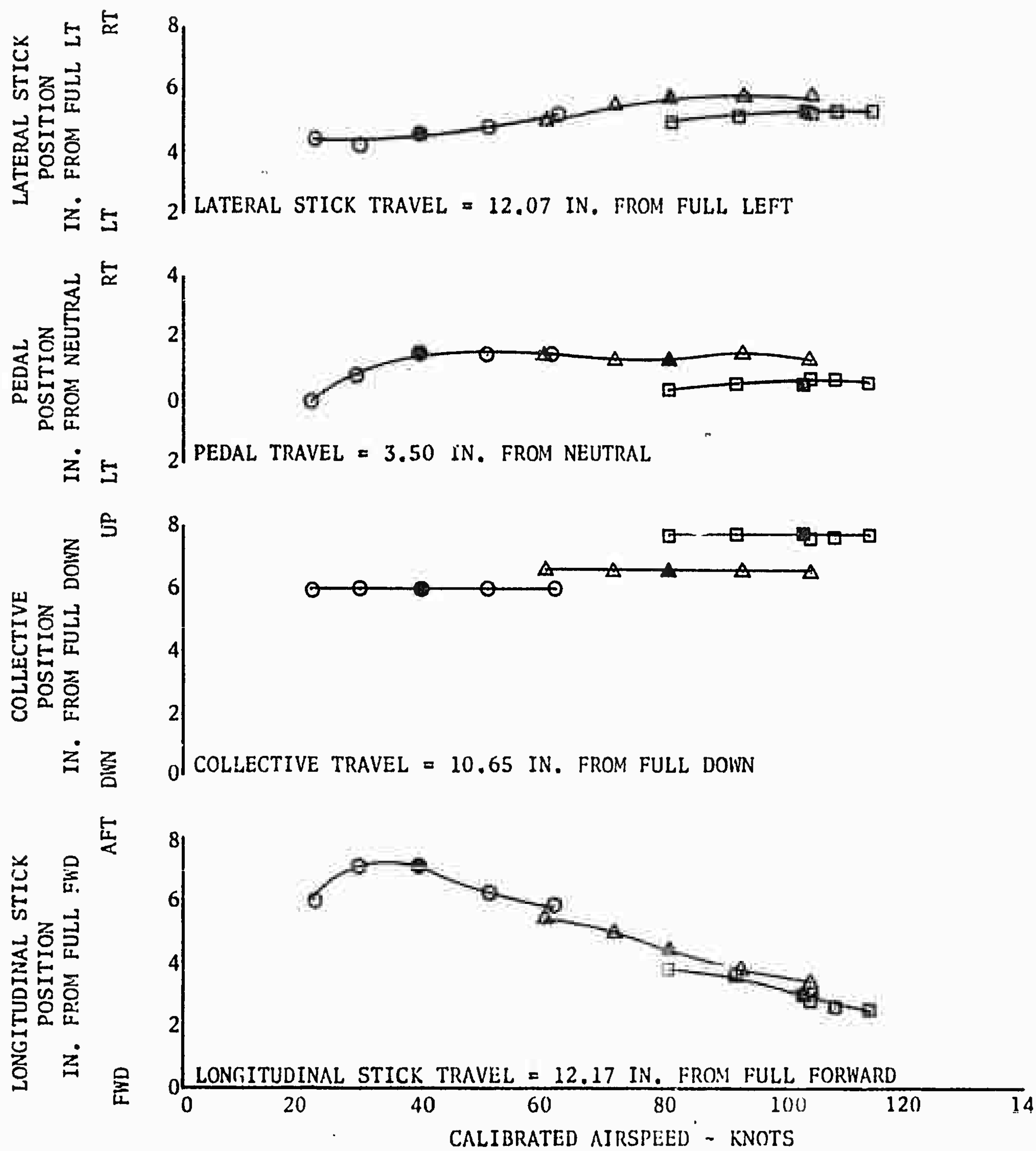


FIGURE NO. 25
STATIC LONGITUDINAL SPEED STABILITY
UH-1B/540 USA S/N 64-14105
XM-16/M-5 ARMAMENT SUBSYSTEM

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	GROSS WEIGHT LBS
○	39	5180	126.7	324	9511
△	76.5	5450	126.6	324	9426
□	93.5	5780	126.4	324	9285

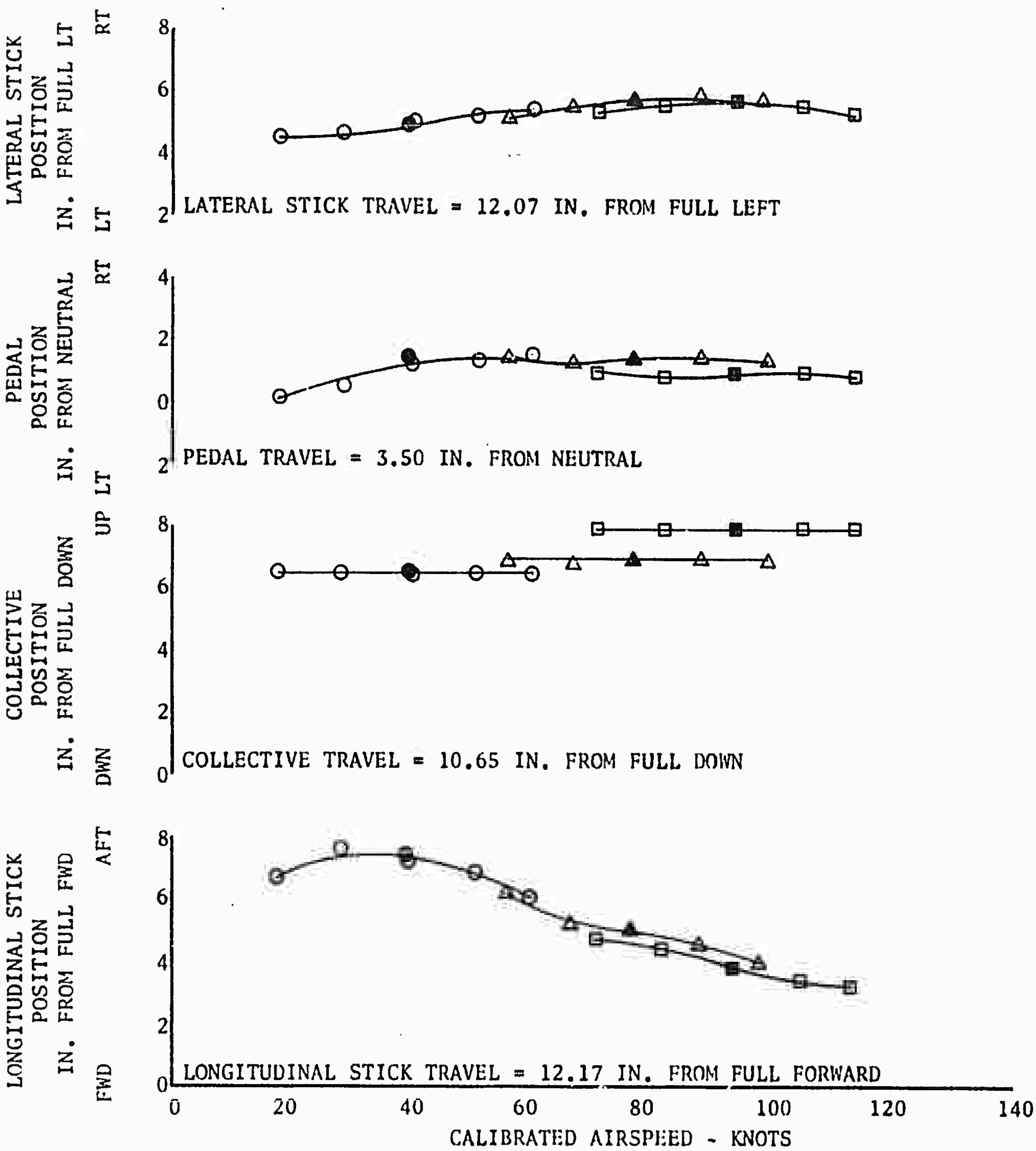


FIGURE NO. 36
STATIC LONGITUDINAL SPEED STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

SYM	TRIM AIRSPEED KCAS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	GROSS WEIGHT LBS
○	39	5600	126.2	324	7593
△	88.5	5020	126.3	324	7683
□	110	4100	126.6	324	7909

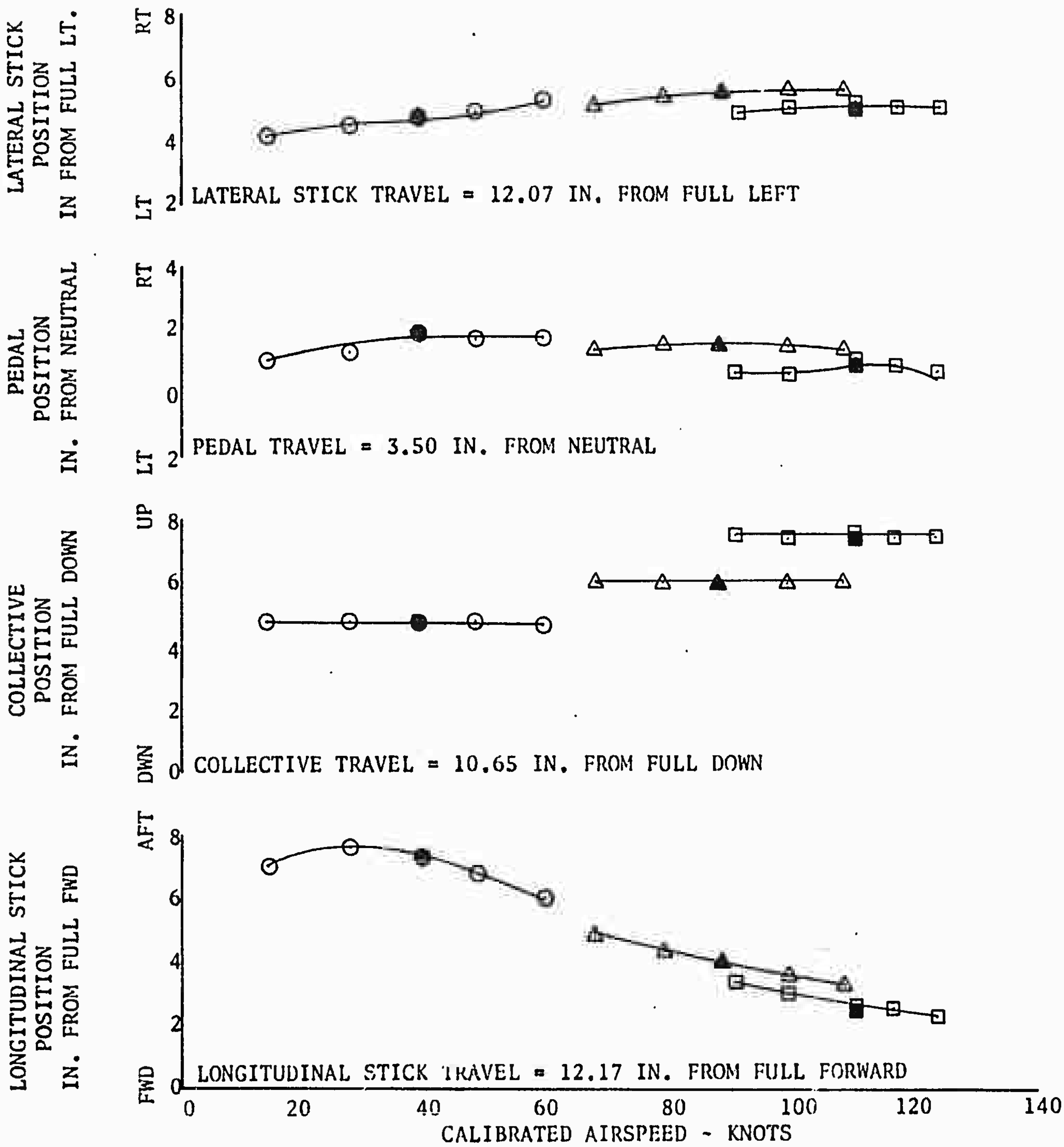


FIGURE NO. 27
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7885 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.5 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 53.5 KTS

SYM		
△	ROLL ANGLE	Ø
□	PITCH ANGLE	Θ

LEVEL FLIGHT

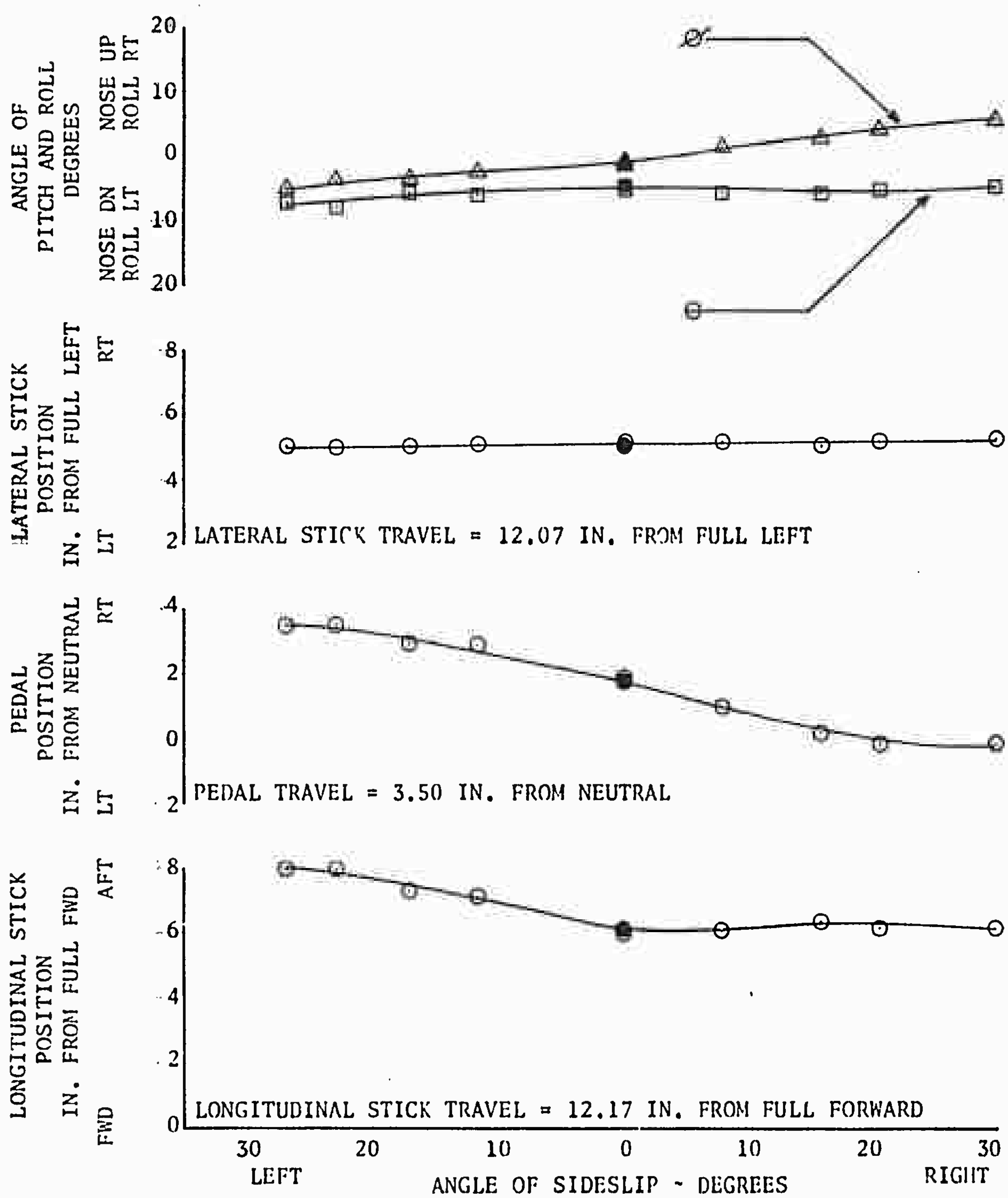


FIGURE NO. 28
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
LEVEL FLIGHT

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9425	5000	324	126.5 (FWD)	56	XM-16/M-5
○	9370	5000	324	126.6 (FWD)	55	XM-21/M-5

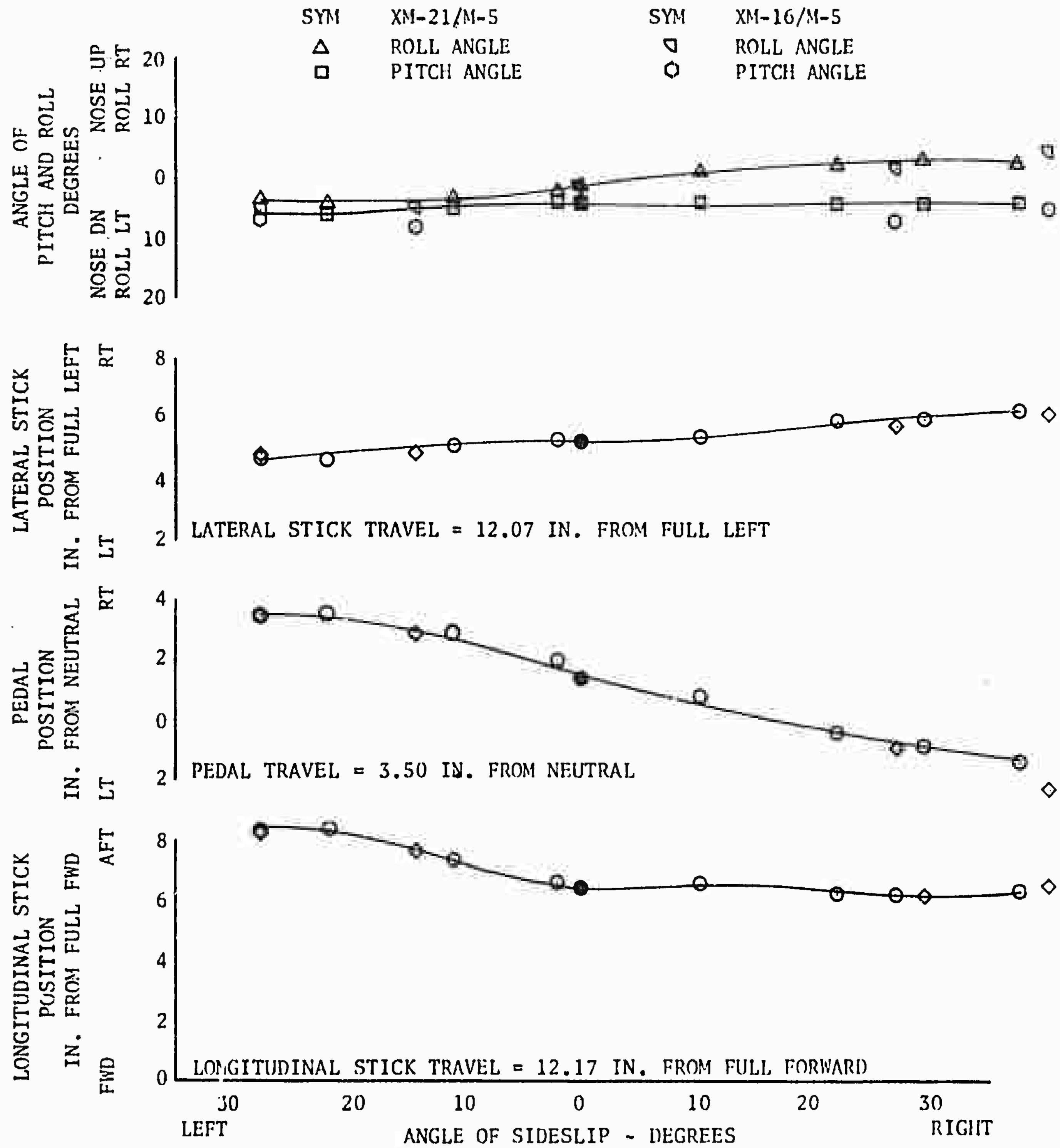


FIGURE NO. 29
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7820 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 85 KTS

SYM
△ ROLL ANGLE Ø
□ PITCH ANGLE ⊖

LEVEL FLIGHT

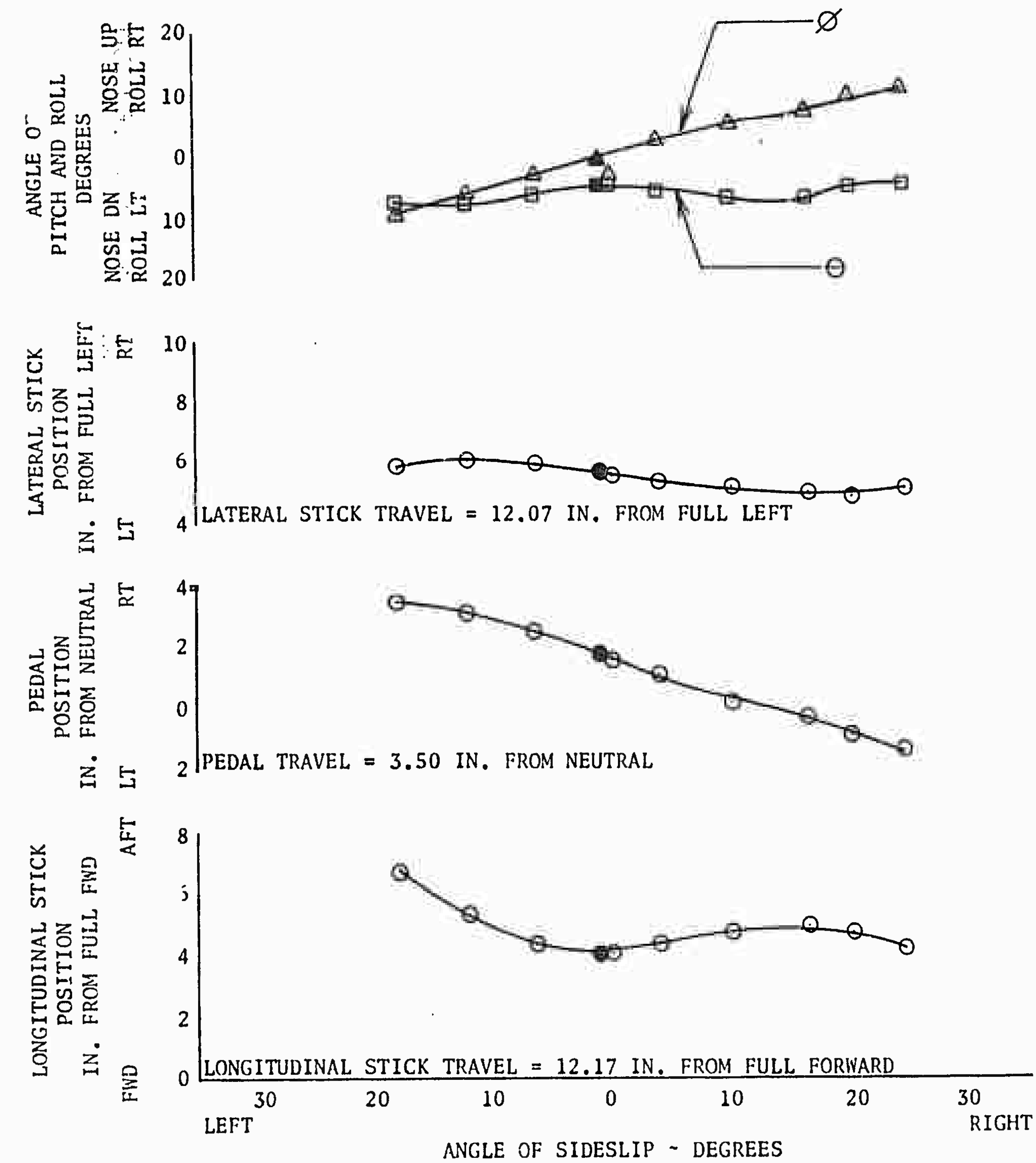


FIGURE NO. 30
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
LEVEL FLIGHT

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9385	5000	324	126.5 (FWD)	76.5	XM-16/M-5
○	9100	5000	324	126.4 (FWD)	82	XM-21/M-5

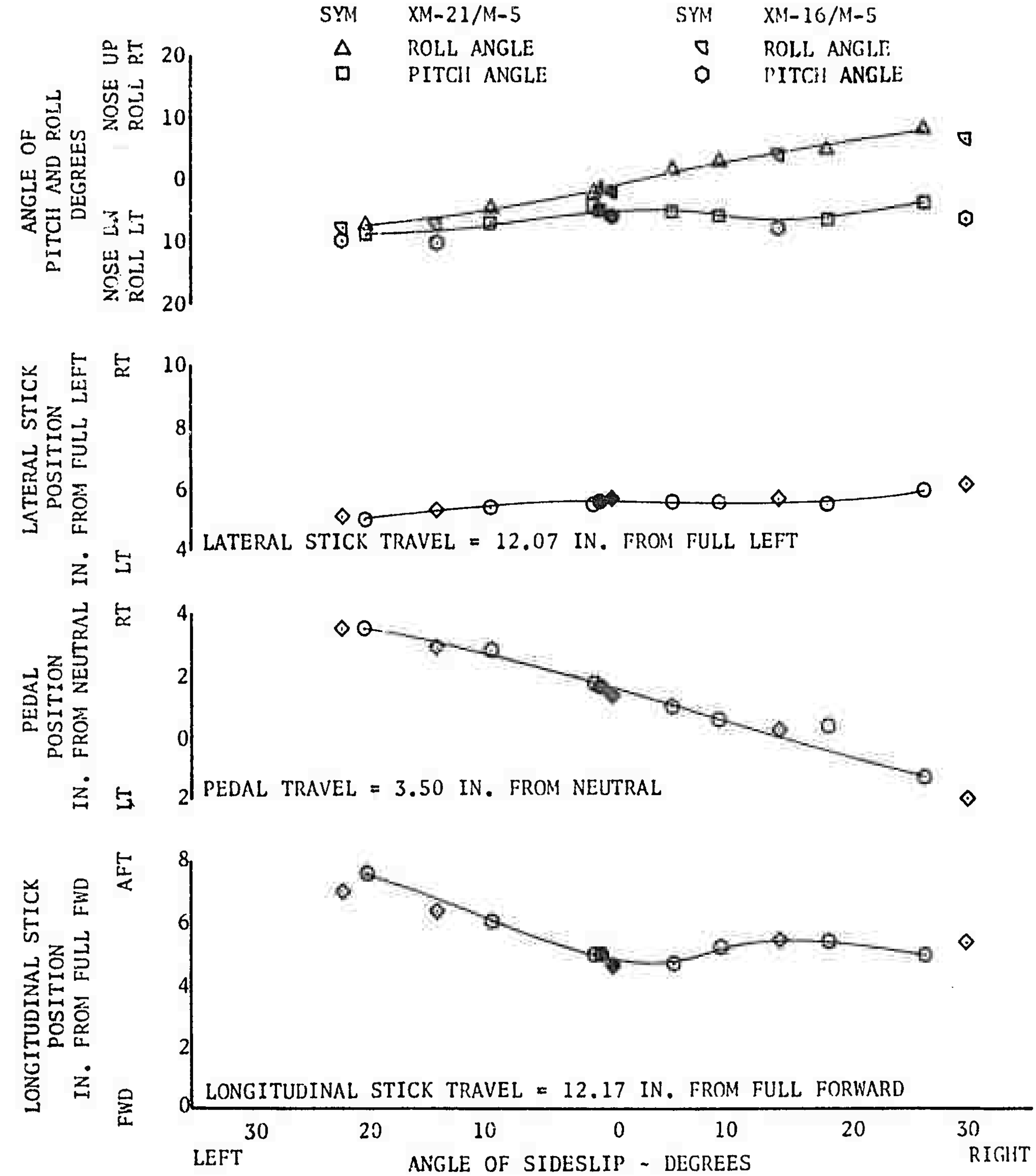


FIGURE NO. 31

STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7630 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.1 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 105 KTS

SYM	
△	ROLL ANGLE ϕ
□	PITCH ANGLE θ

LEVEL FLIGHT

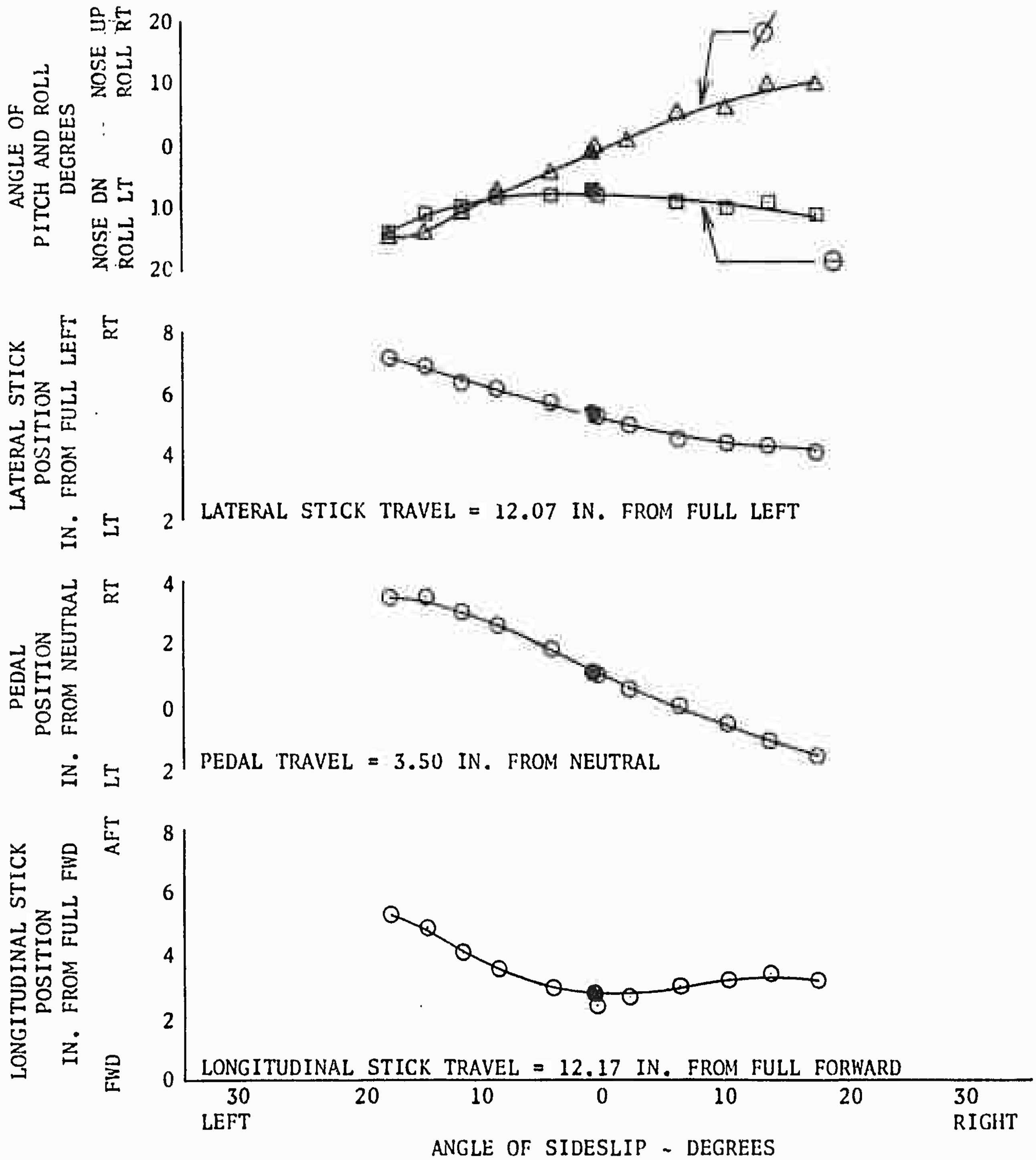


FIGURE NO. 32
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
LEVEL FLIGHT

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9345	5000	324	126.4 (FWD)	94.5	XM-16/M-5
○	9263	5000	324	126.5 (FWD)	102	XM-21/M-5

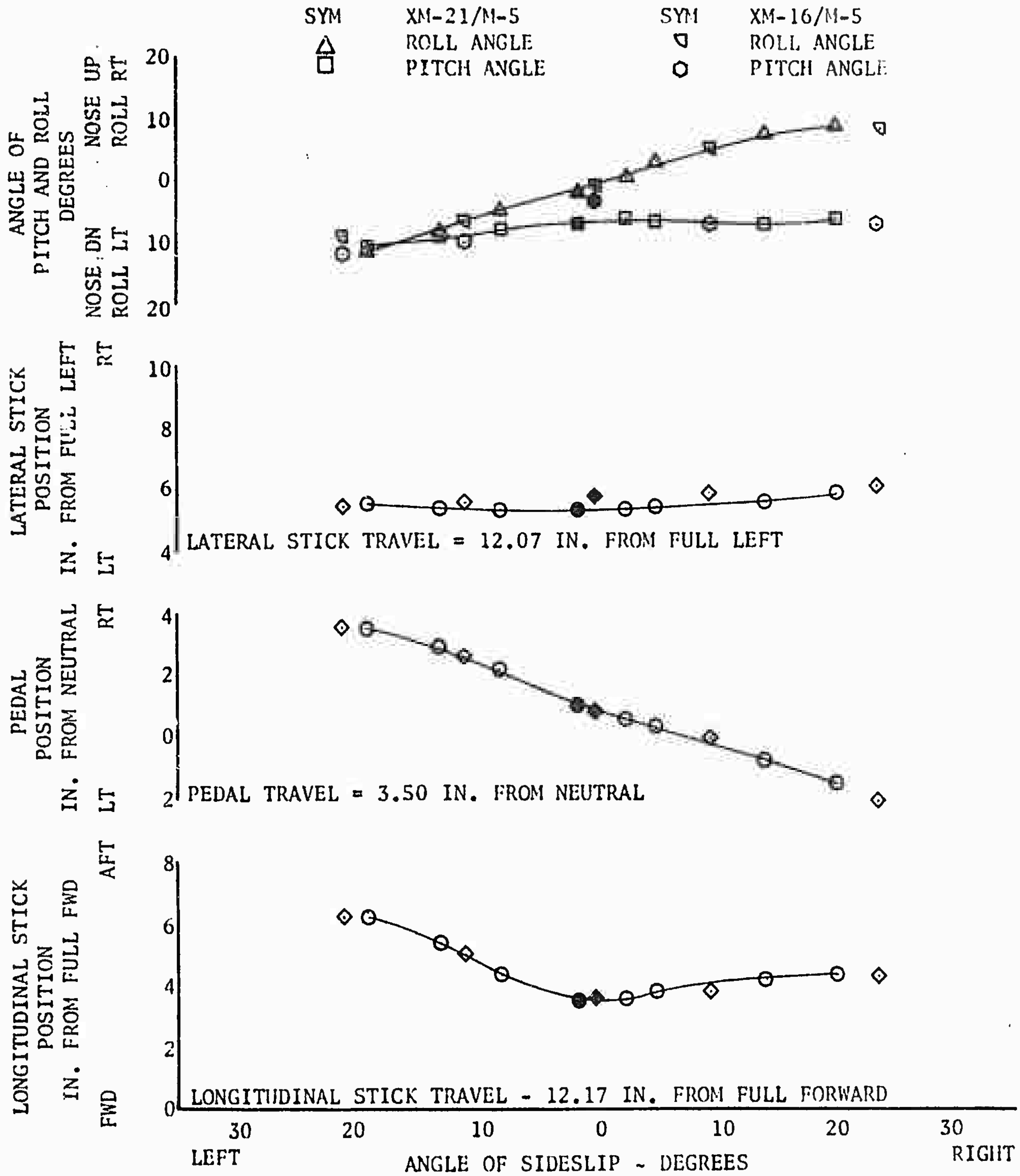


FIGURE NO. 33
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7715 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.2 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 123 KTS

POWERED DESCENT

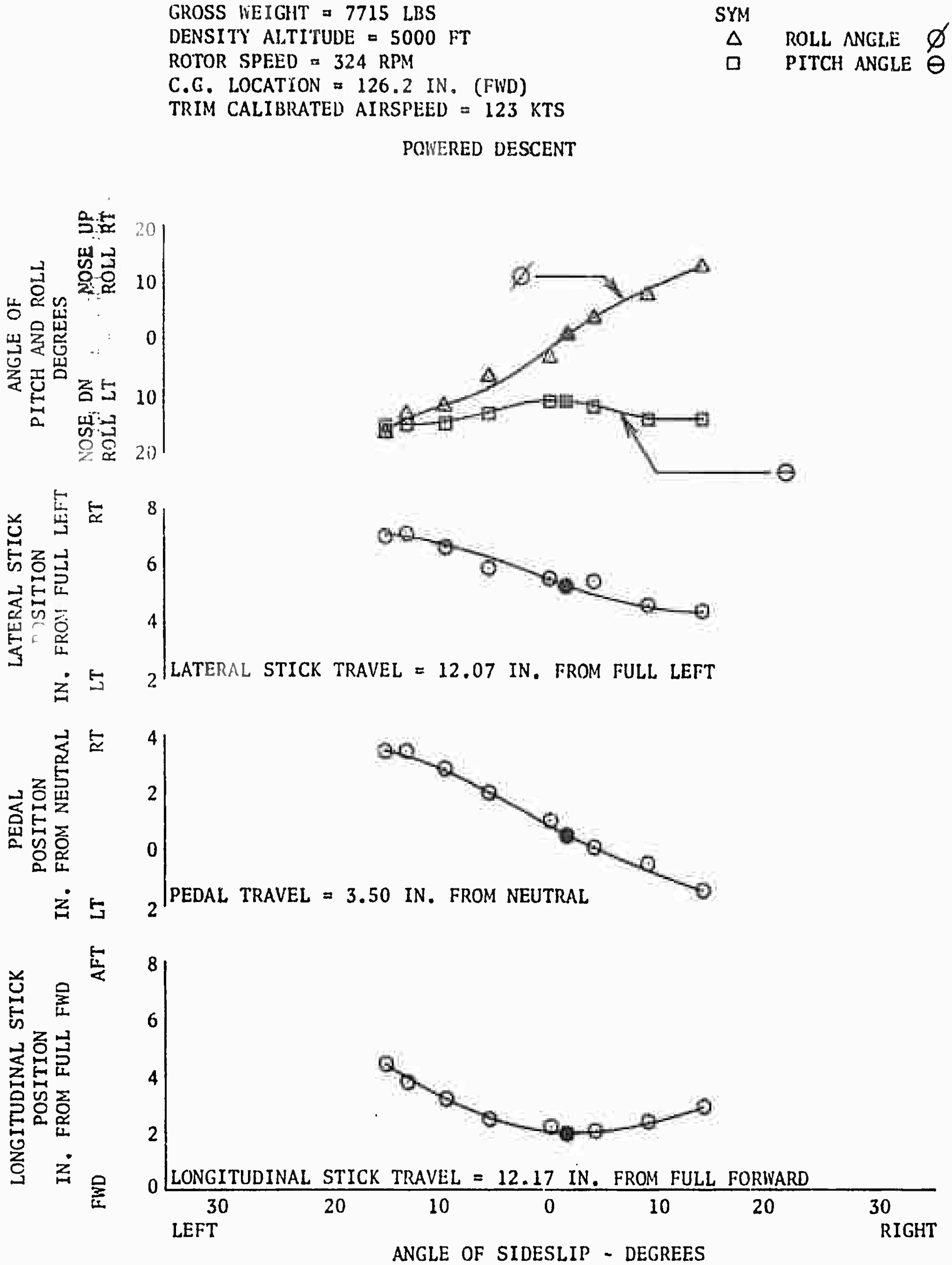


FIGURE NO. 34
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
POWERED DESCENT

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9270	5000	324	126.3 (FWD)	118	XM-16/M-5
○	9225	5000	324	126.5 (FWD)	113.5	XM-21/M-5

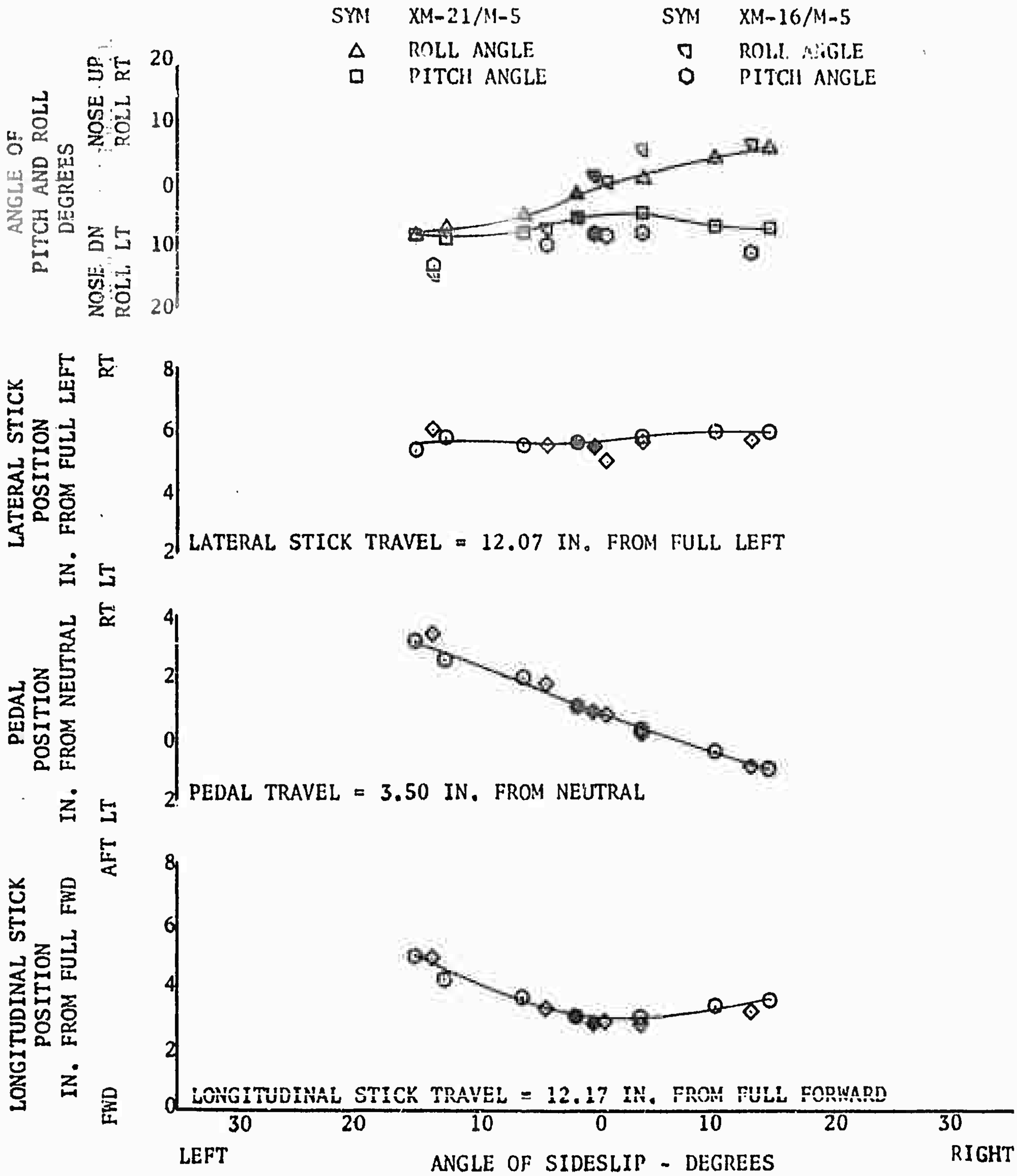


FIGURE NO. 35
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

CLIMB

GROSS WEIGHT = 7870 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 51 KTS

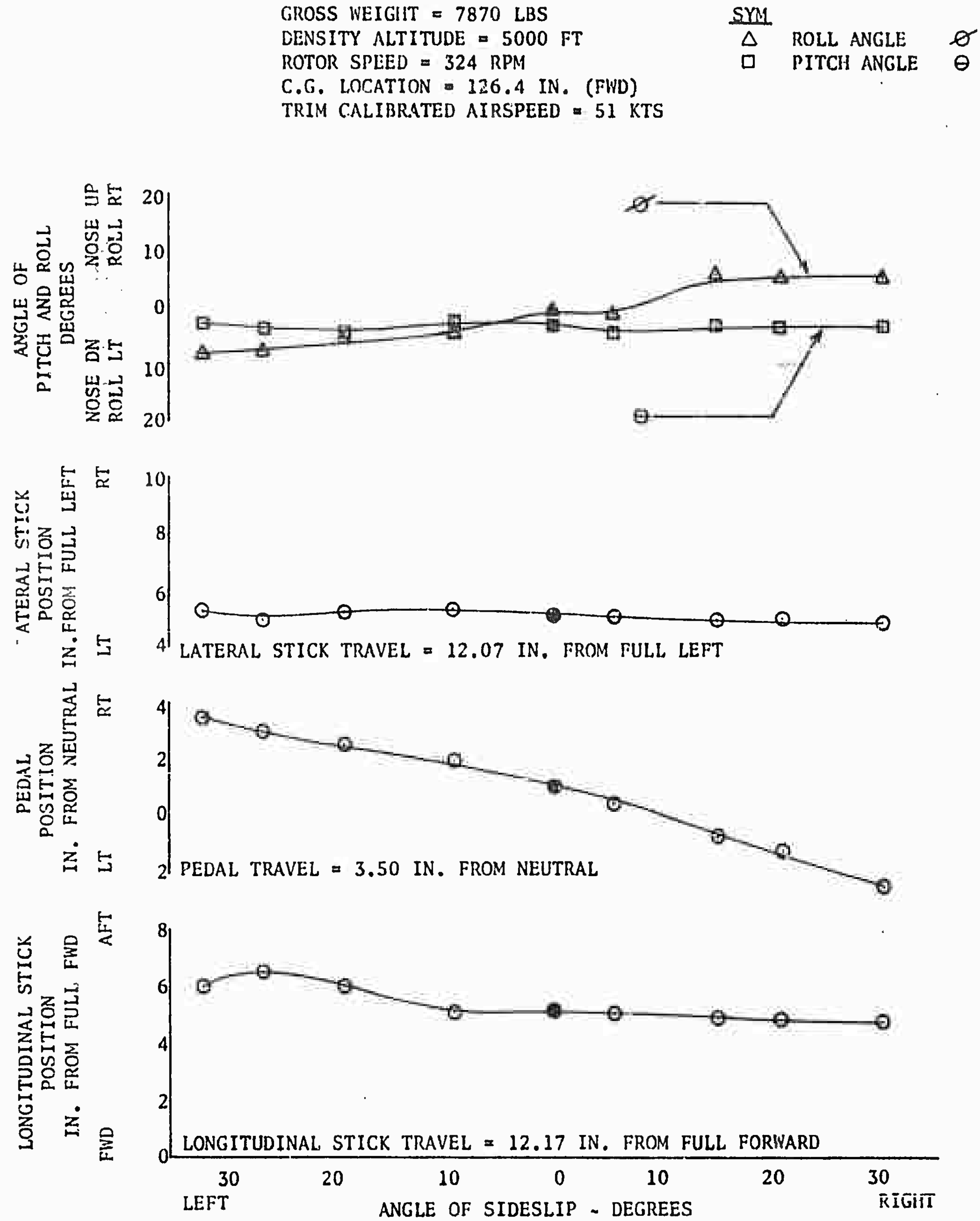


FIGURE NO. 36
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
CLIMB

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9425	5000	324	126.5 (FWD)	56	XM-16/M-5
○	9075	5000	324	126.3 (FWD)	56	XM-21/M-5

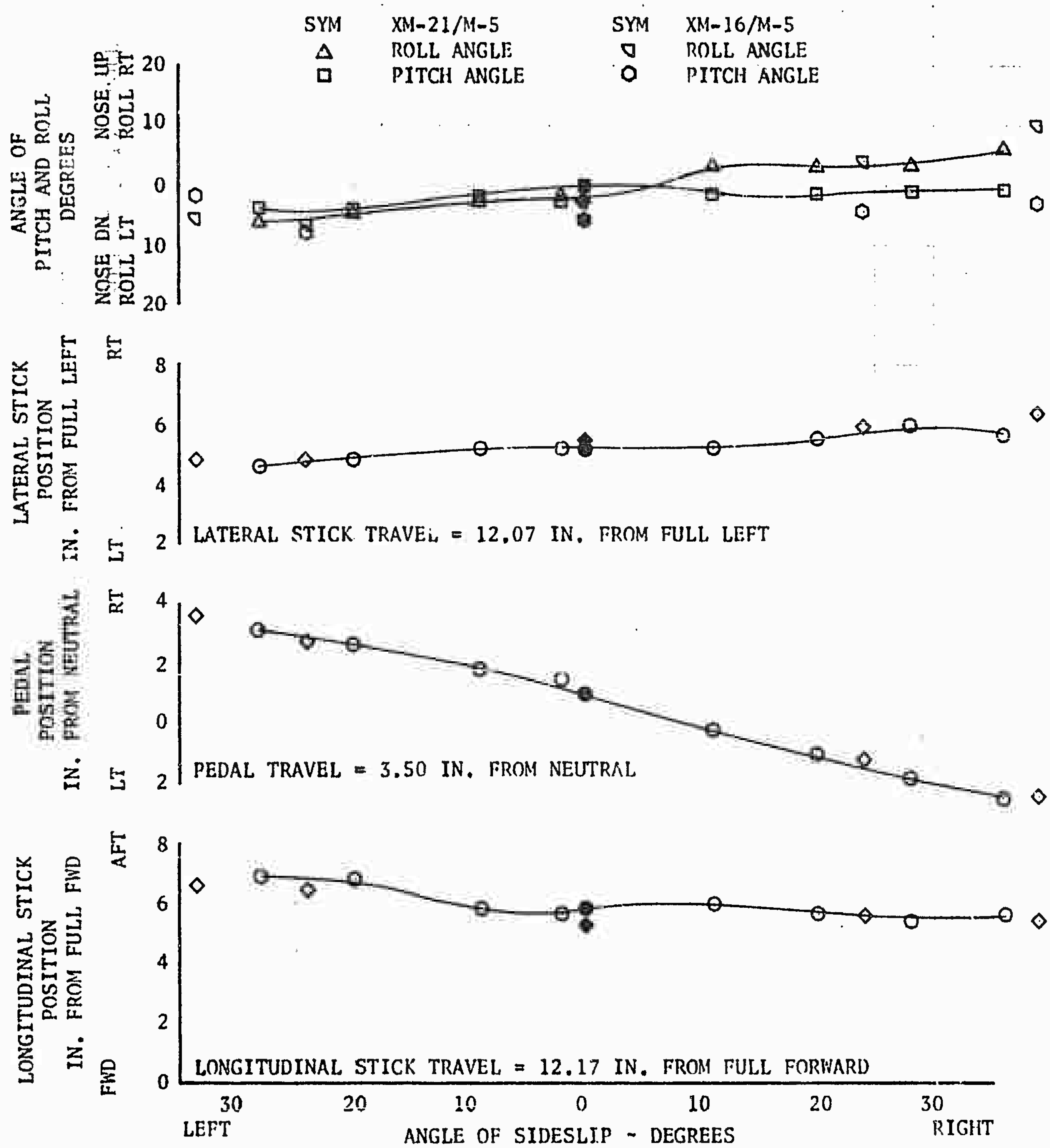


FIGURE NO. 37
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7840 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN (FWD)
TRIM CALIBRATED AIRSPEED = 57 KTS

AUTOROTATION

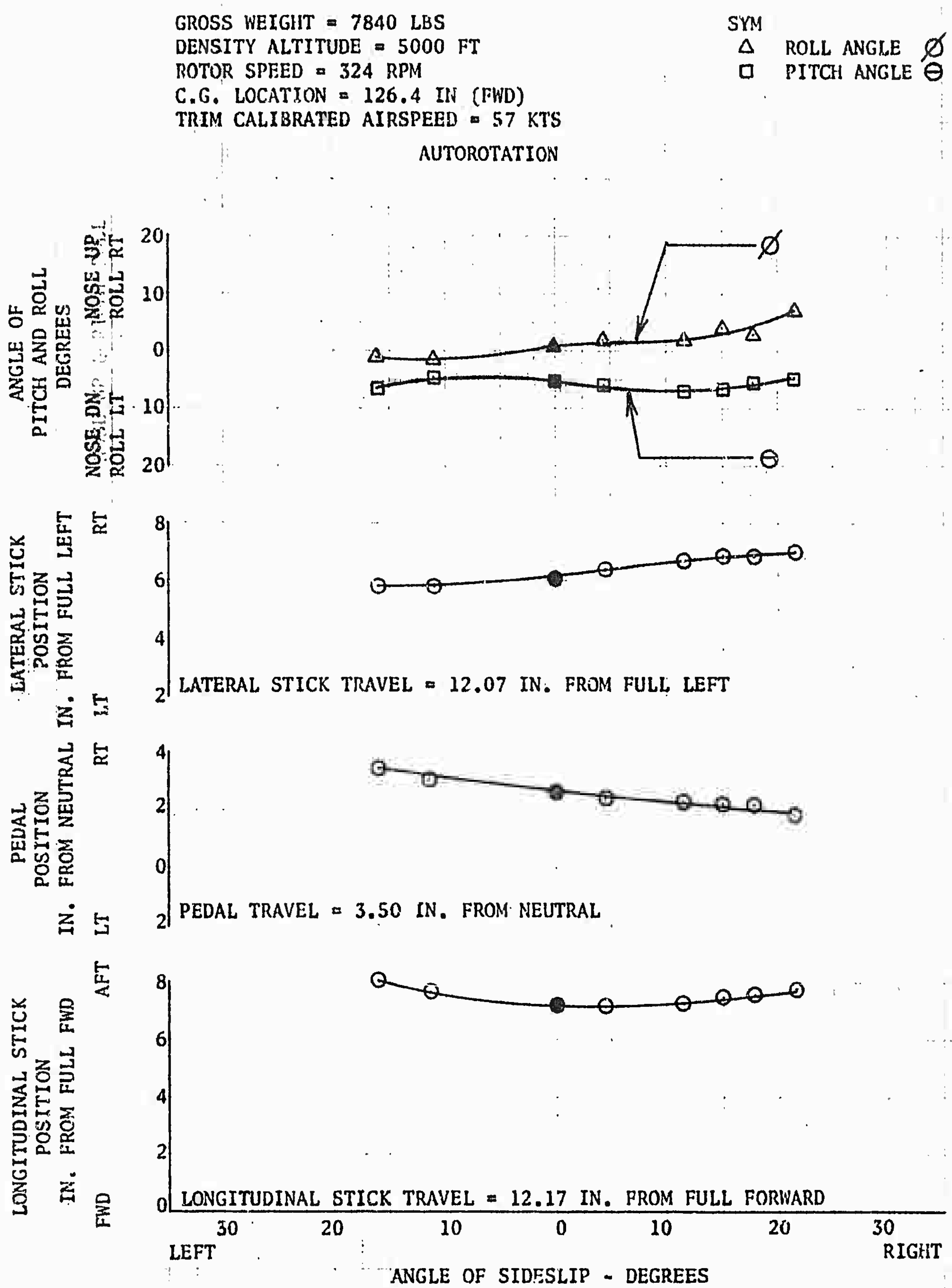


FIGURE NO. 38
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
AUTOROTATION

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	ROTOR SPEED RPM	C.G. LOCATION IN	TRIM CALIBRATED AIRSPEED KTS	ARMAMENT SUBSYSTEM
◇	9105	5000	324	126.2 (FWD)	61	XM-16/M-5
○	9310	5000	324	126.6 (FWD)	61	XM-21/M-5

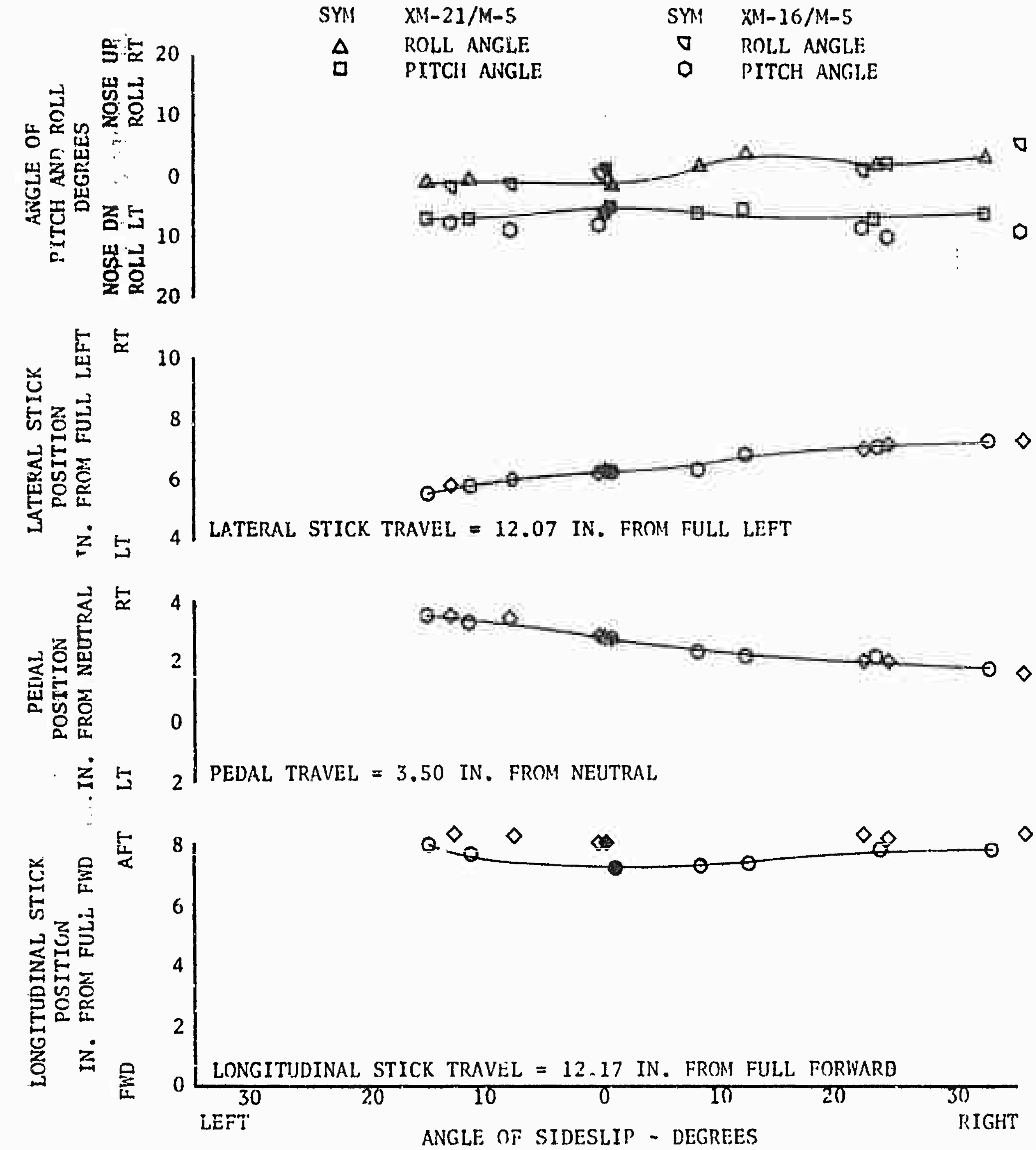


FIGURE NO. 39
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7840 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.8 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 56 KTS

LEVEL FLIGHT

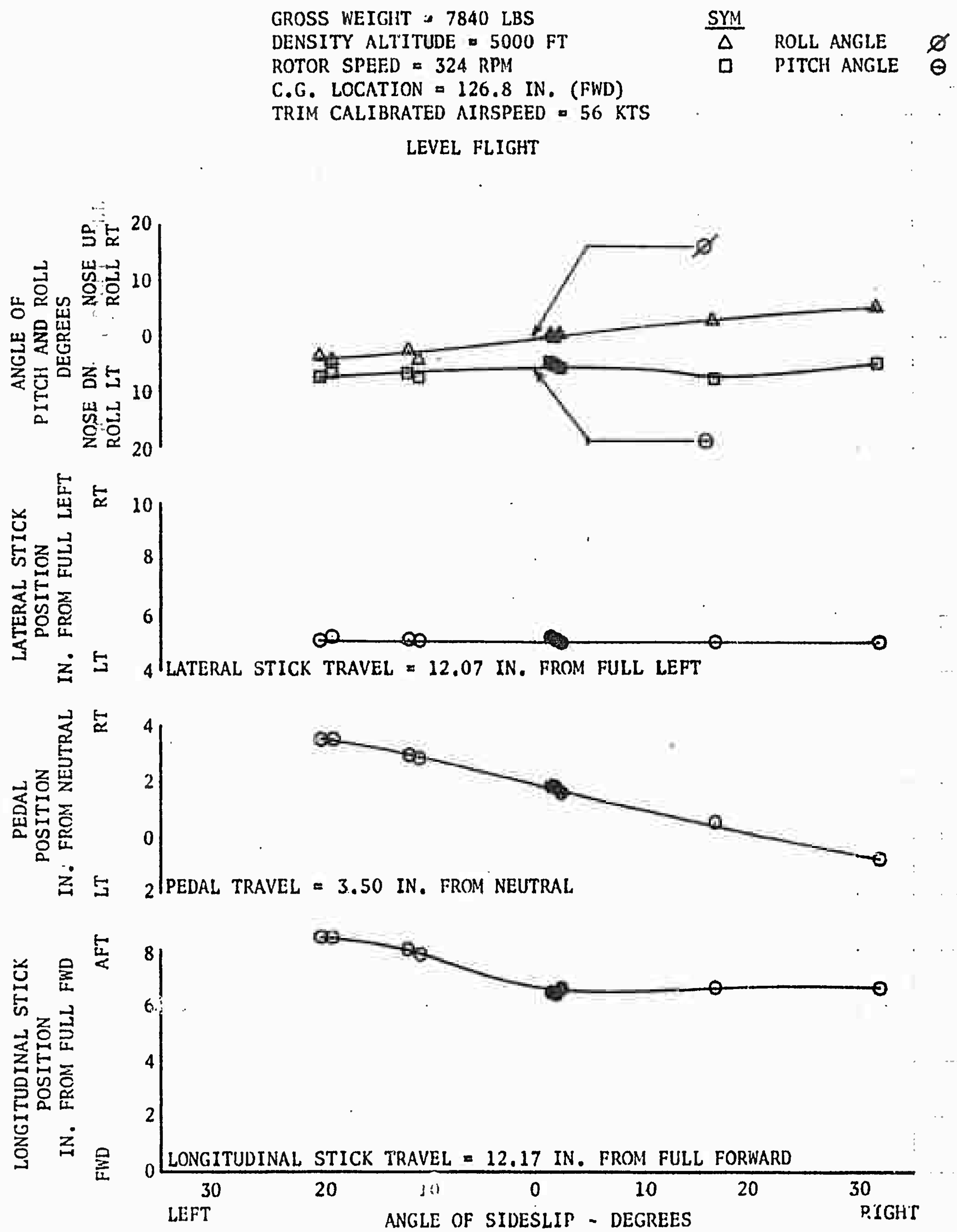


FIGURE NO. 40
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7635 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.6 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 88.5 KTS

LEVEL FLIGHT

SYM	
△	ROLL ANGLE Ø
□	PITCH ANGLE Θ

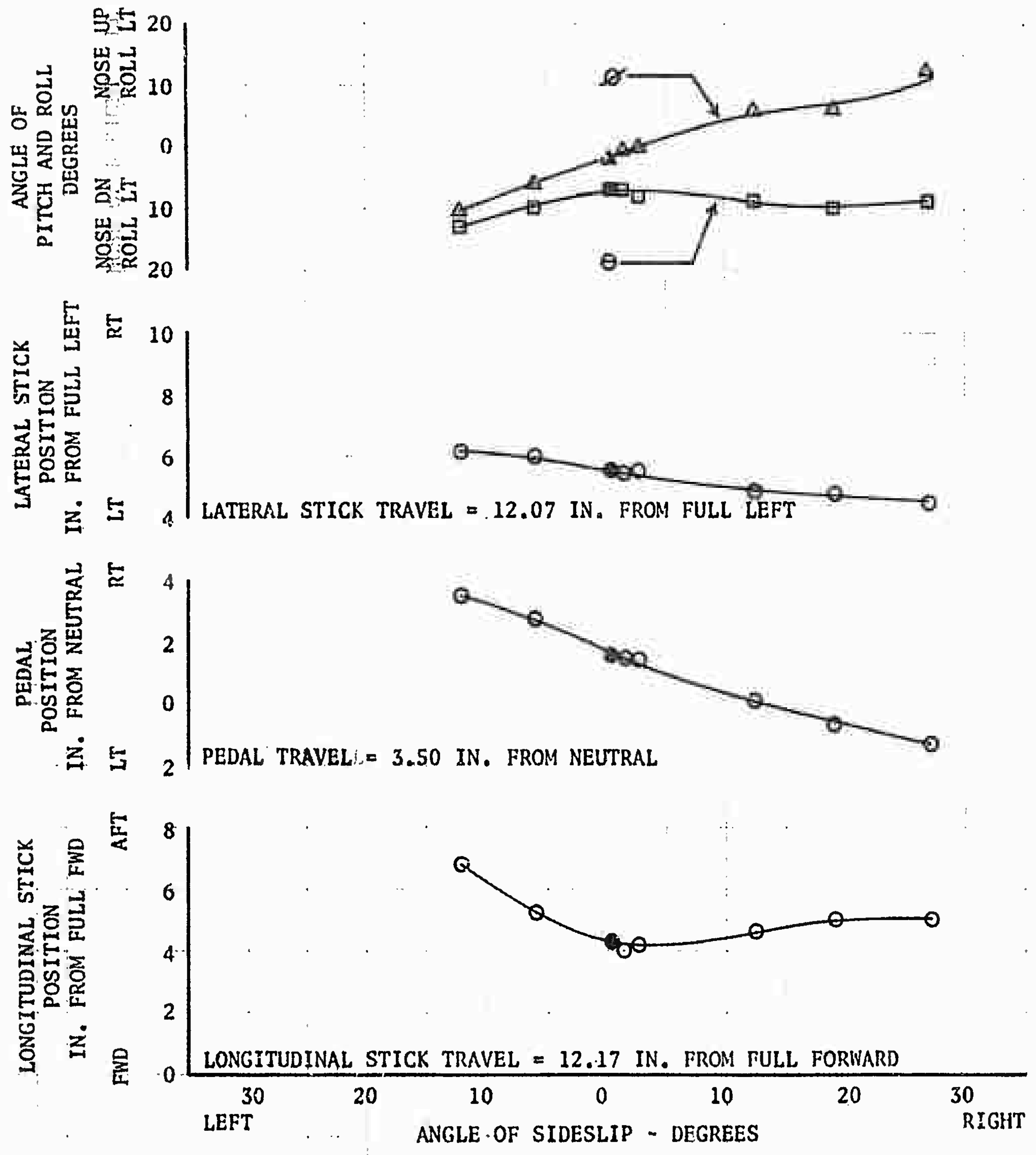


FIGURE NO. 41
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

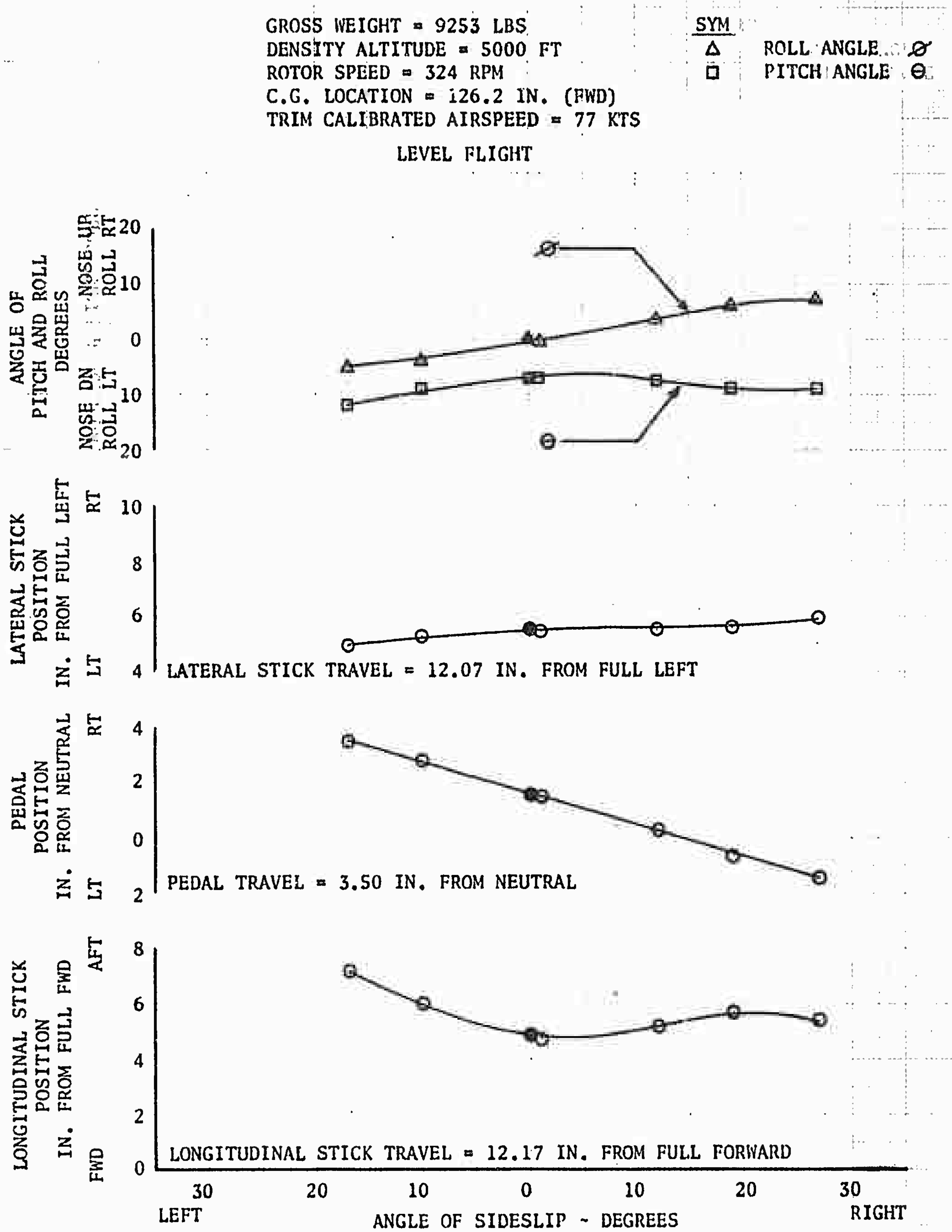


FIGURE NO.42
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7890 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.9 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 107 KTS
LEVEL FLIGHT

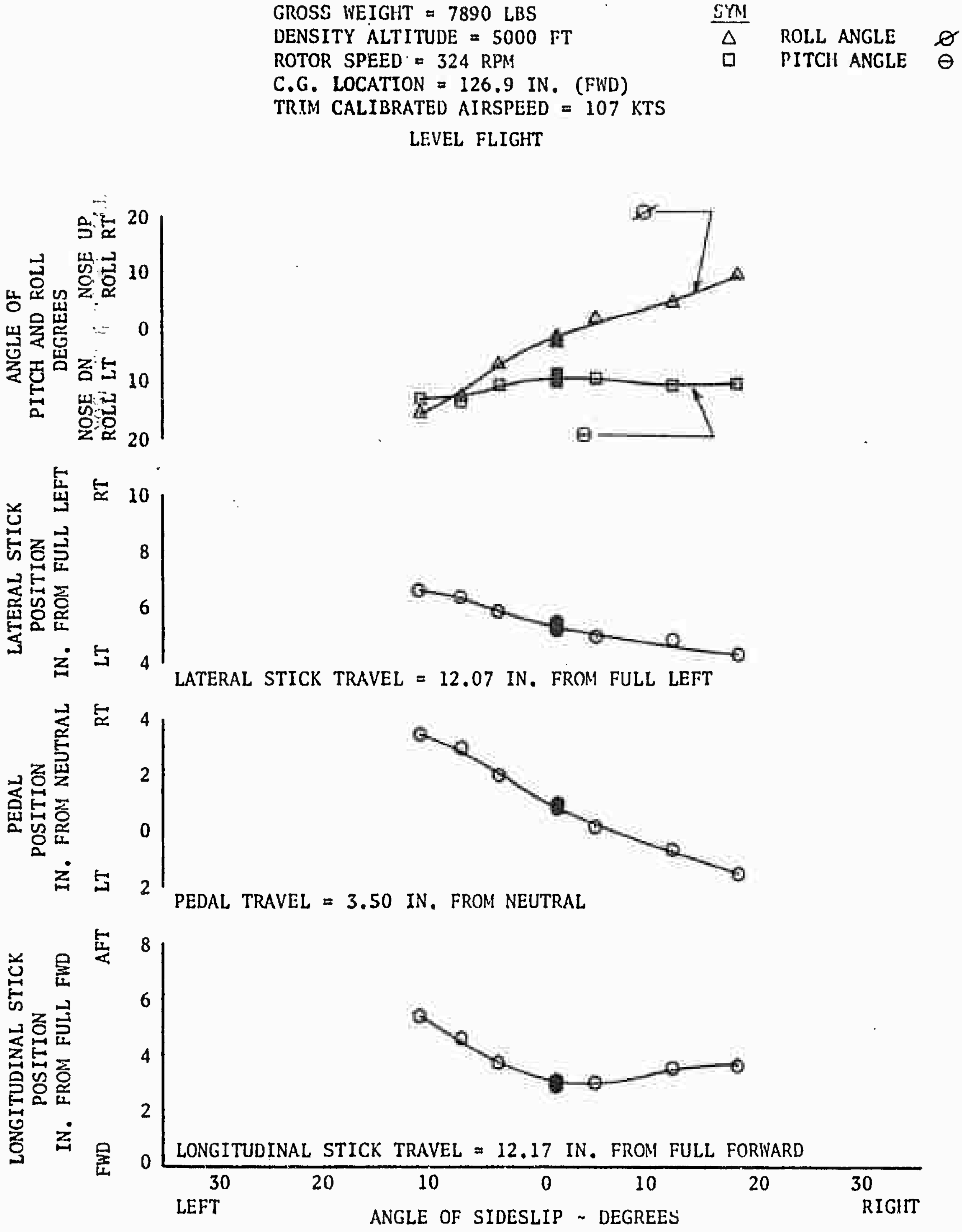


FIGURE NO.43
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 9410 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.4 IN (FWD)
TRIM CALIBRATED AIRSPEED = 96.5 KTS

SYM
△ ROLL ANGLE Ø
□ PITCH ANGLE ⊖

LEVEL FLIGHT

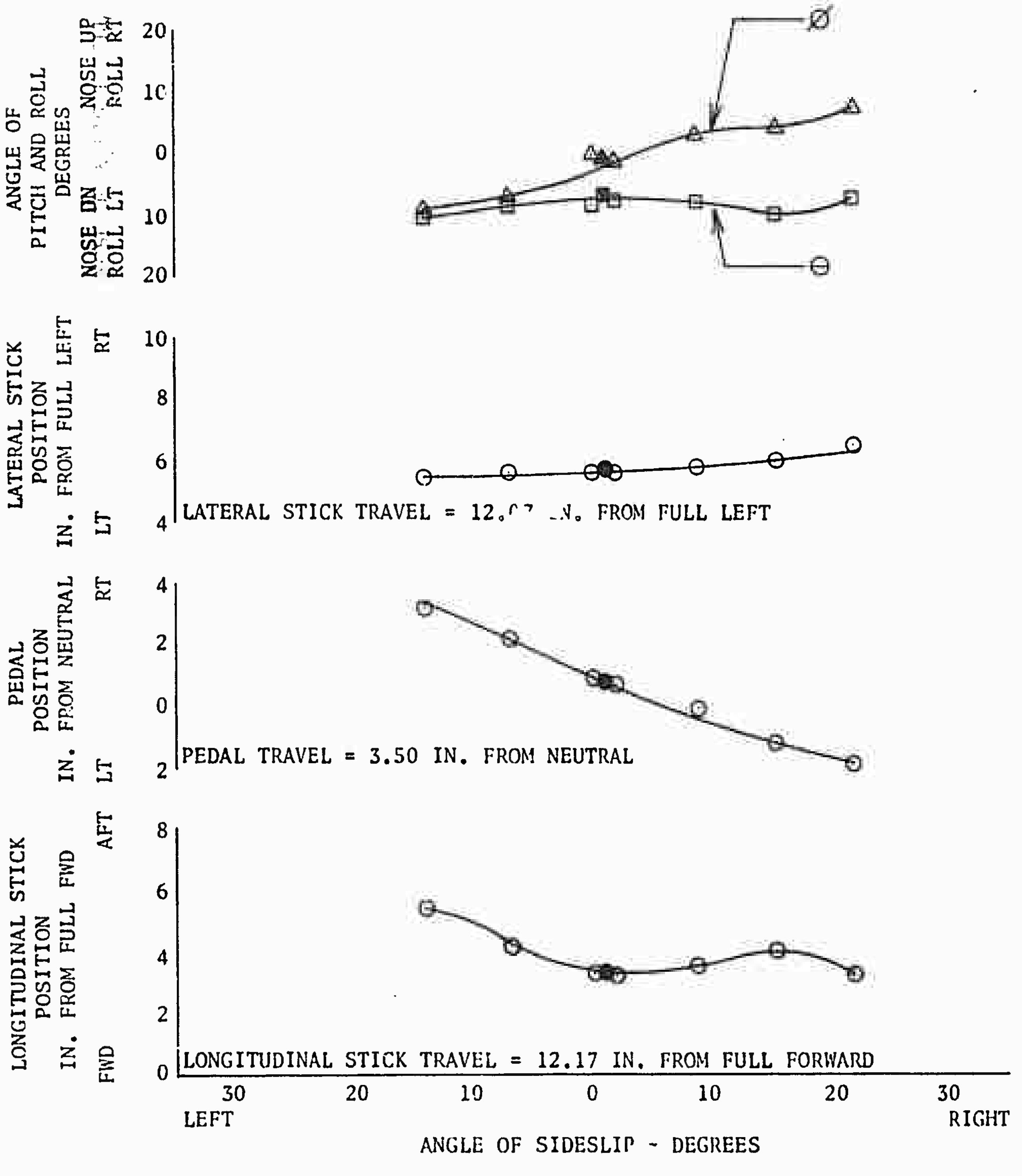


FIGURE NO. 44
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7720 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.7 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 122.5 KTS

SYM
△ ROLL ANGLE Ø
□ PITCH ANGLE Θ

POWERED DESCENT

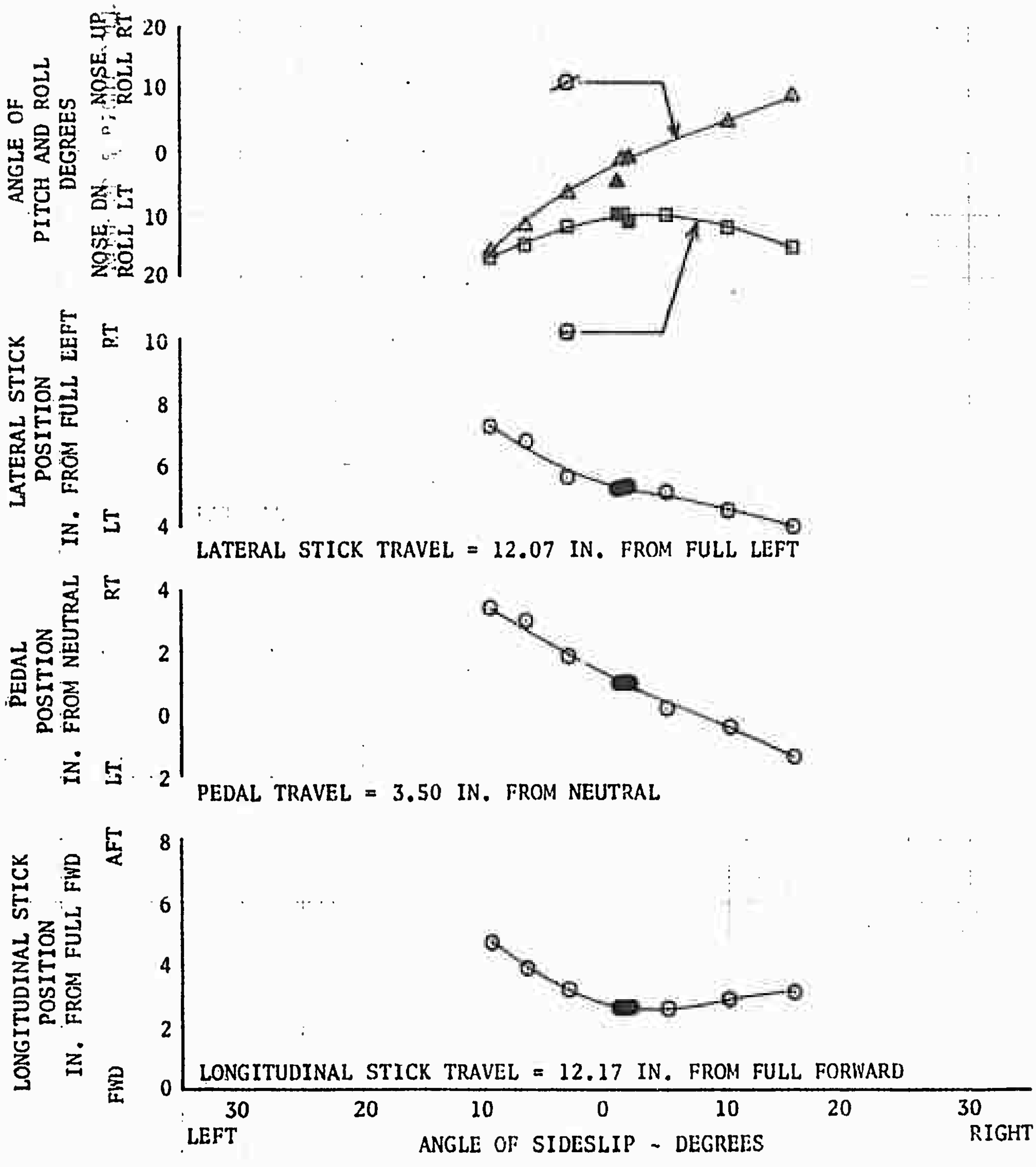


FIGURE NO. 45
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

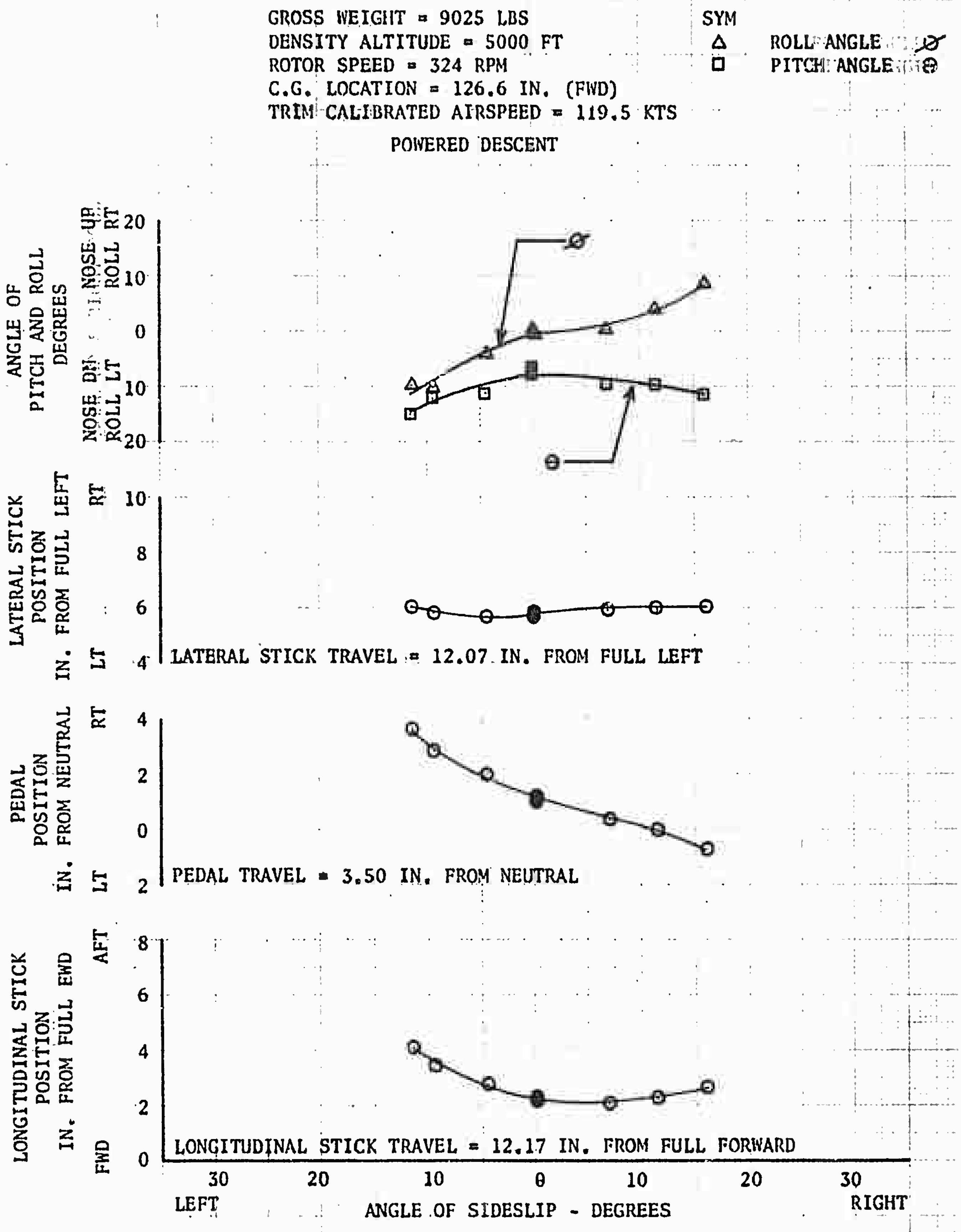


FIGURE NO. 46
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7800 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.6 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 54.5 KTS

SYM
△ ROLL ANGLE Ø
□ PITCH ANGLE θ

CLIMB

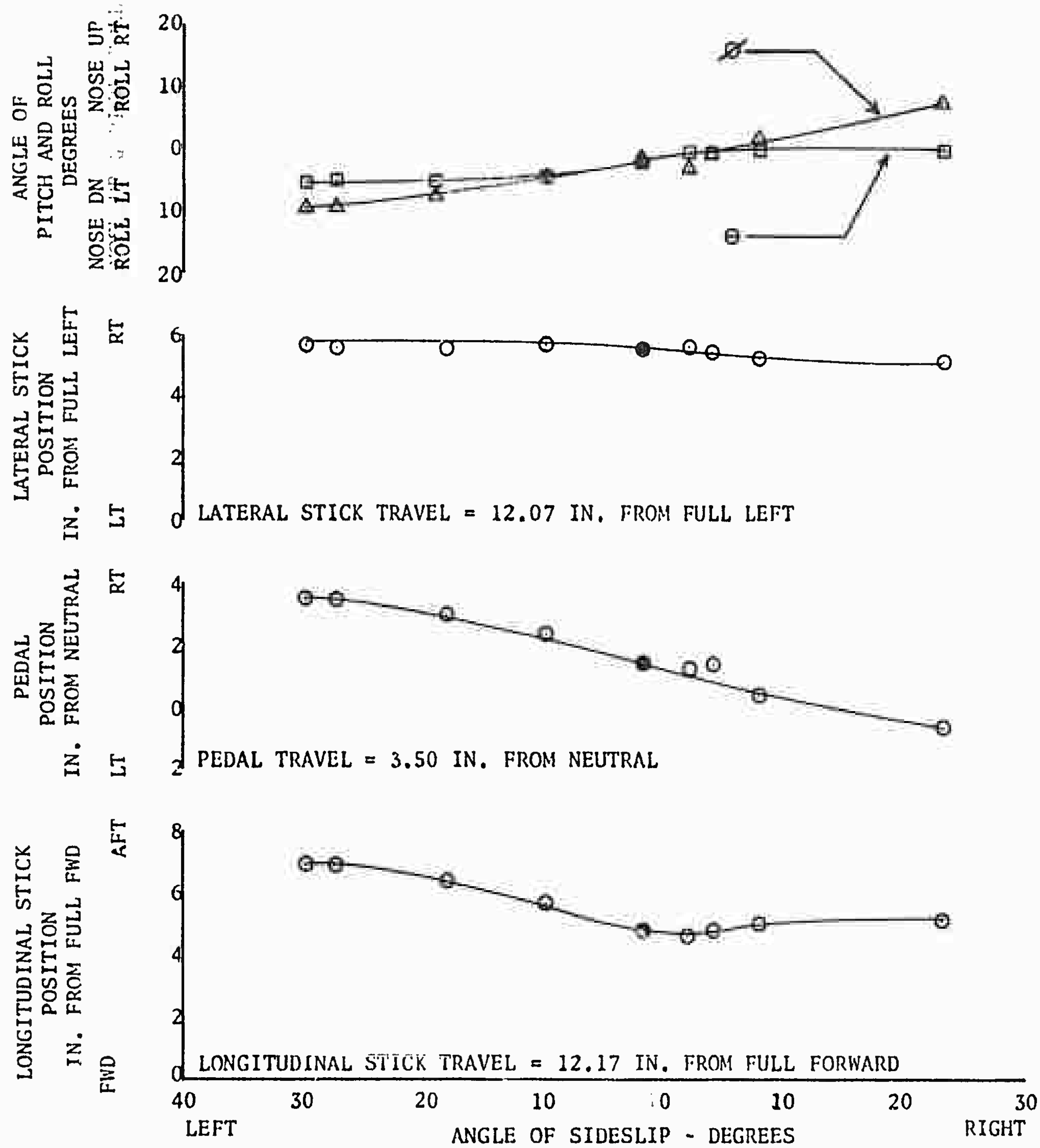


FIGURE NO. 47
STATIC LATERAL DIRECTIONAL STABILITY
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 7846 LBS
DENSITY ALTITUDE = 5000 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 126.6 IN. (FWD)
TRIM CALIBRATED AIRSPEED = 59.5 KTS

AUTOROTATION

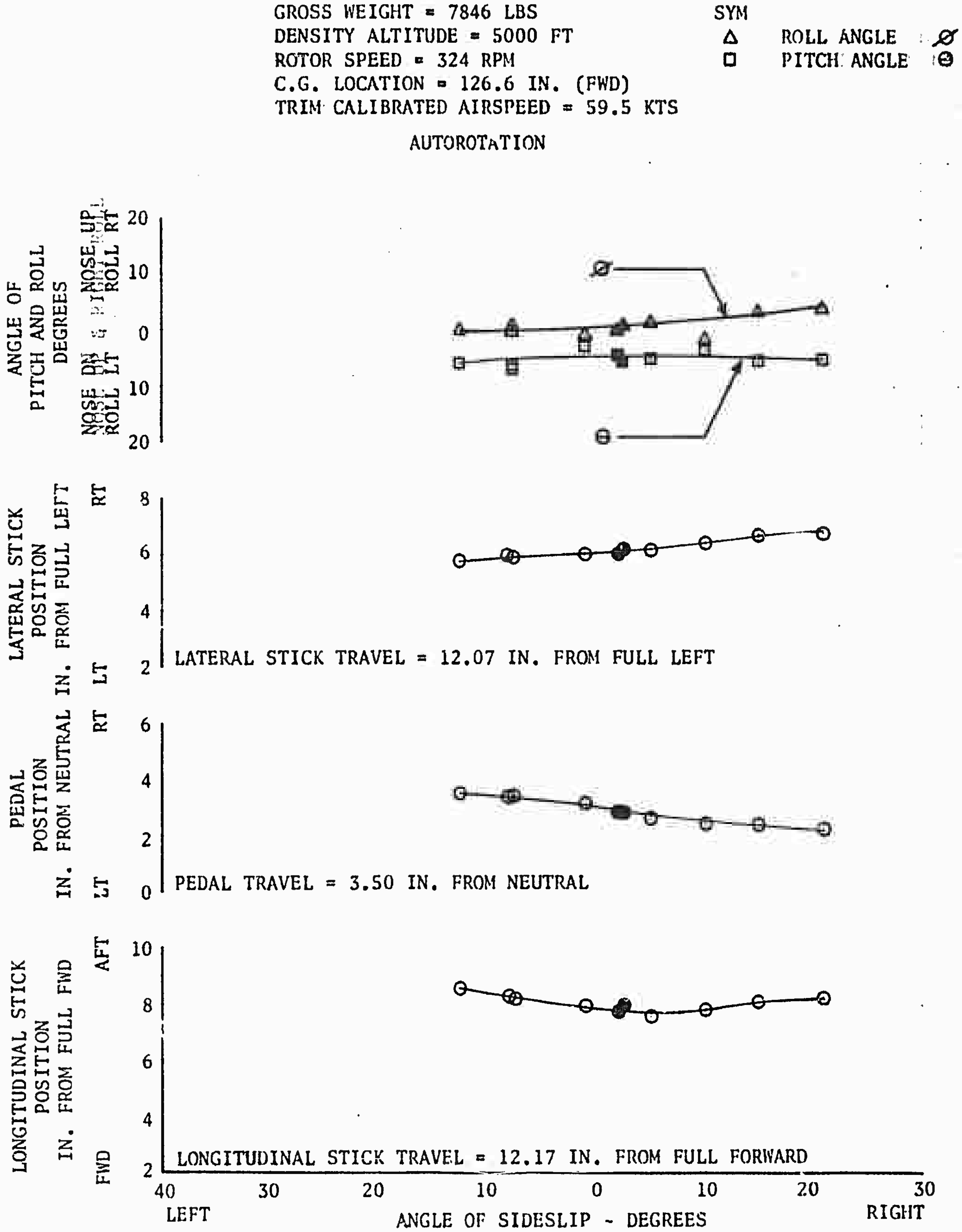


FIGURE NO. 48
CONTROL POSITION IN SIDEWARD FLIGHT
UH-1B/540 USA S/N 64-14105
XM-16/M-5 ARMAMENT SUBSYSTEM

GROSS WEIGHT = 8300 LBS
DENSITY ALTITUDE = 740 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 127.8 (FWD)

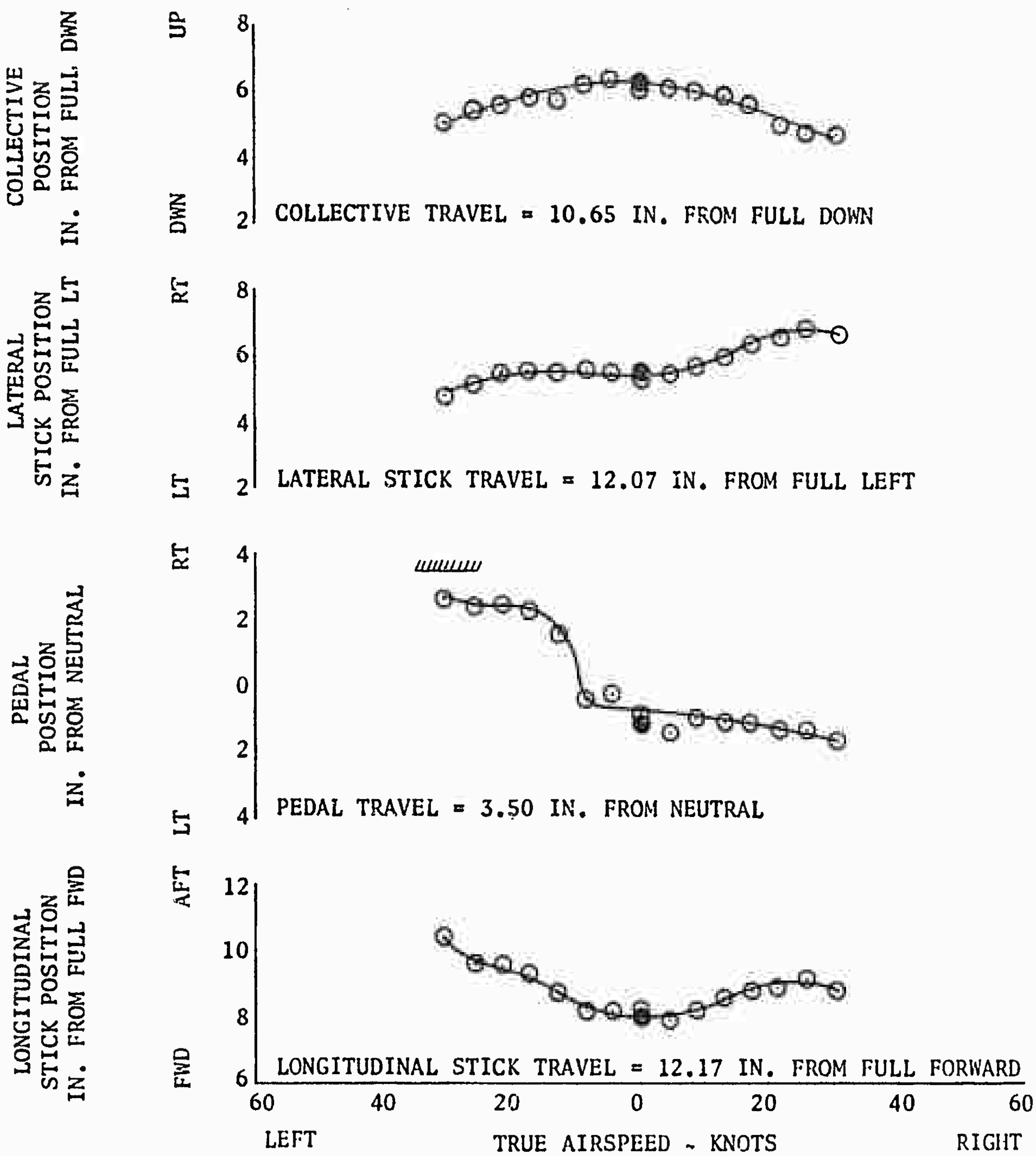


FIGURE NO. 49
CONTROL POSITION IN REARWARD AND FORWARD FLIGHT
UH-1B/540 USA S/N 64-14105
XM-16/M-5 ARMAMENT SUBSYSTEM
GROSS WEIGHT = 8100 LBS
DENSITY ALTITUDE = 740 FT
ROTOR SPEED = 324 RPM
C.G. LOCATION = 127.5 IN (FWD)

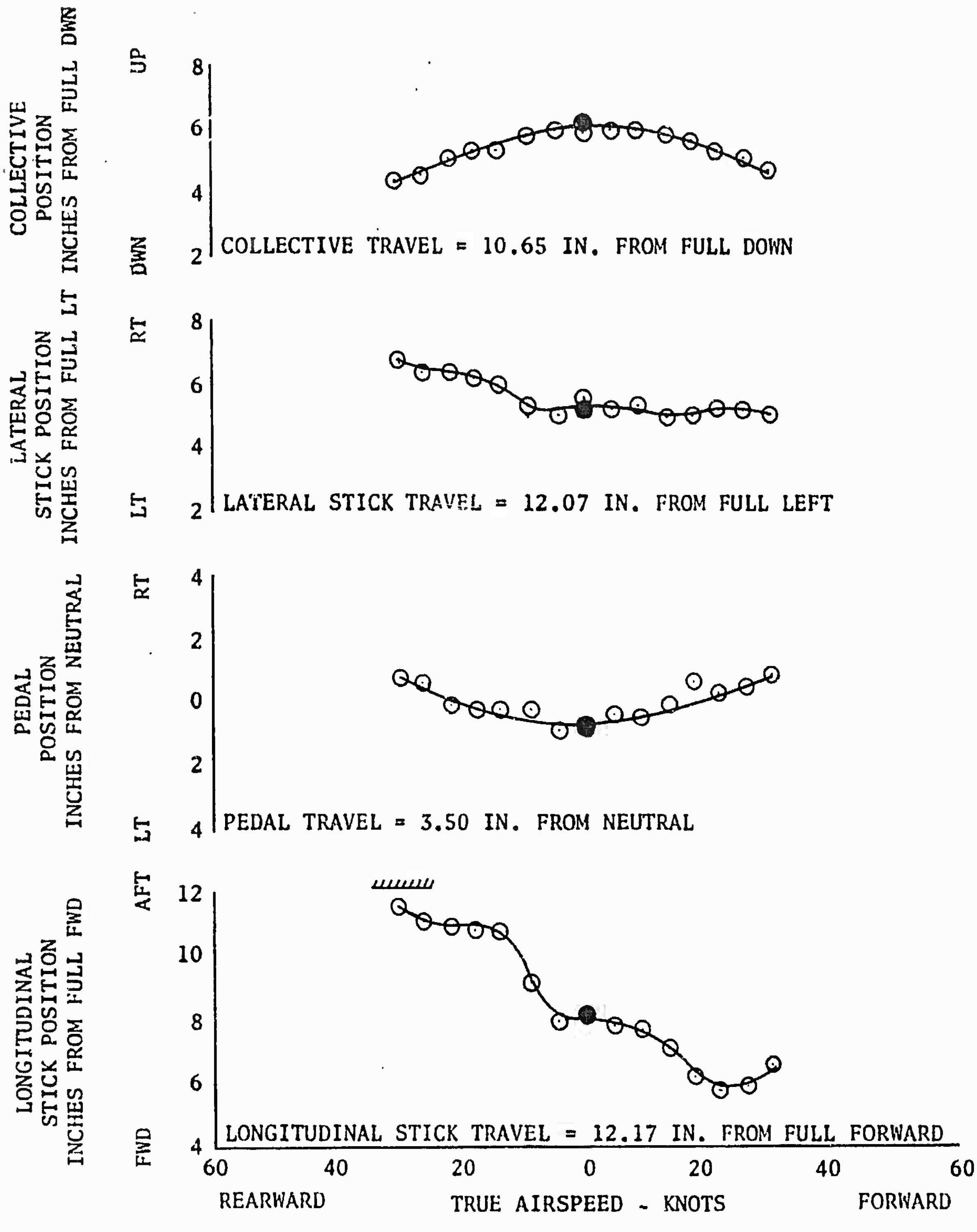


FIGURE NO. 50
SUMMARY OF CONTROL RESPONSE
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
○	7733	5000	126.3	324	XM-21/M-5
⌀	7878	5000	126.5	324	XM-21/M-5
○ (flagged)	7878	5000	126.5	324	XM-21/M-5

NOTES:
1. POINTS OBTAINED FROM FIGURES NO. 56 THROUGH 58
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. FLAGGED SYMBOL DENOTES AUTOROTATION.
4. SLASHED SYMBOL DENOTES CLIMB.
5. CONTROL RESPONSE VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

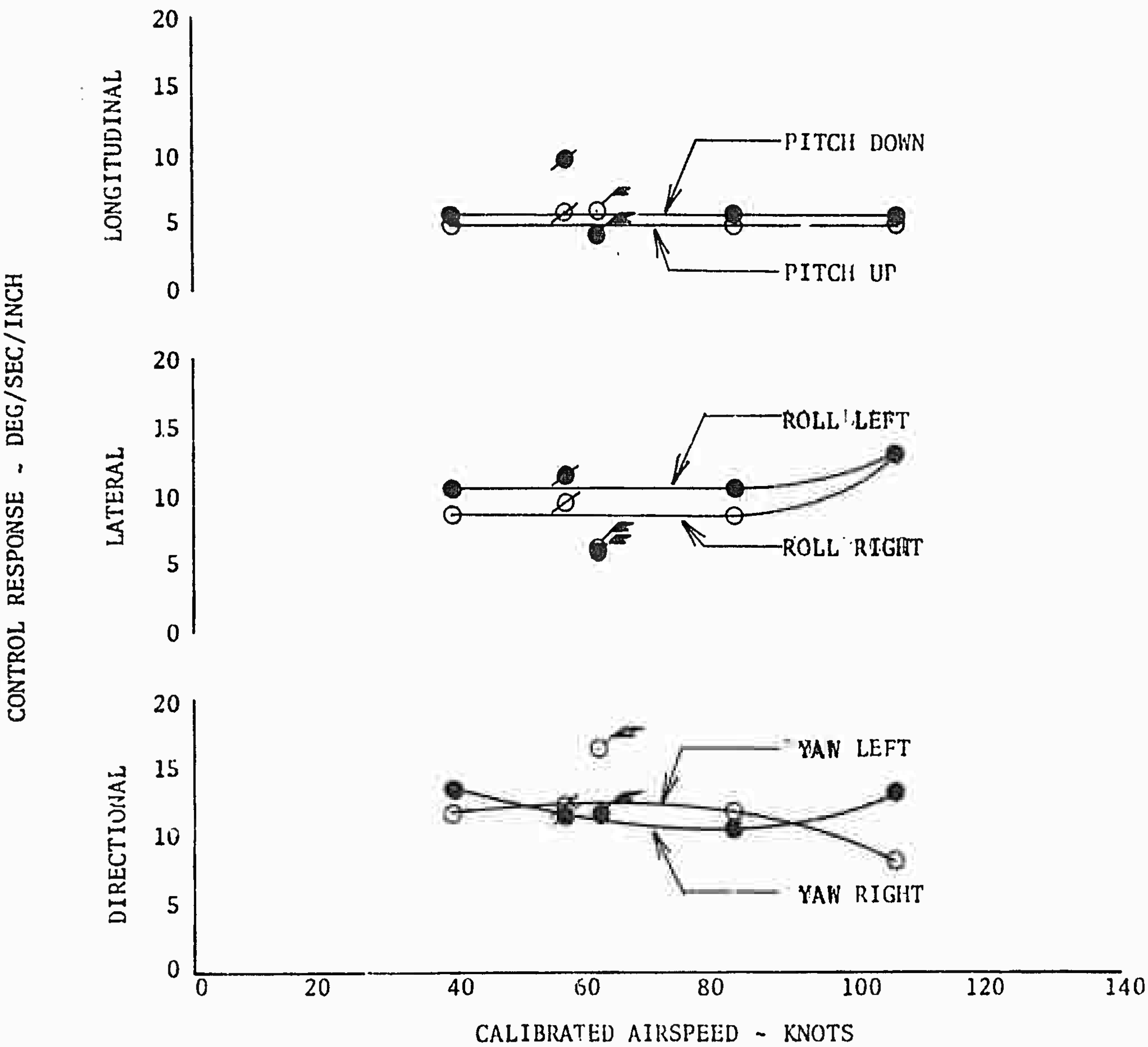


FIGURE NO. 51
SUMMARY OF CONTROL SENSITIVITY
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
○	7733	5000	126.3	324	XM-21/M-5
⌀	7878	5000	126.5	324	XM-21/M-5
○ (flagged)	7878	5000	126.5	324	XM-21/M-5

NOTES:

1. POINTS OBTAINED FROM FIGURES NO. 56 THROUGH 58
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. FLAGGED SYMBOL DENOTES AUTOROTATION.
4. SLASHED SYMBOL DENOTES CLIMB.
5. CONTROL SENSITIVITY VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

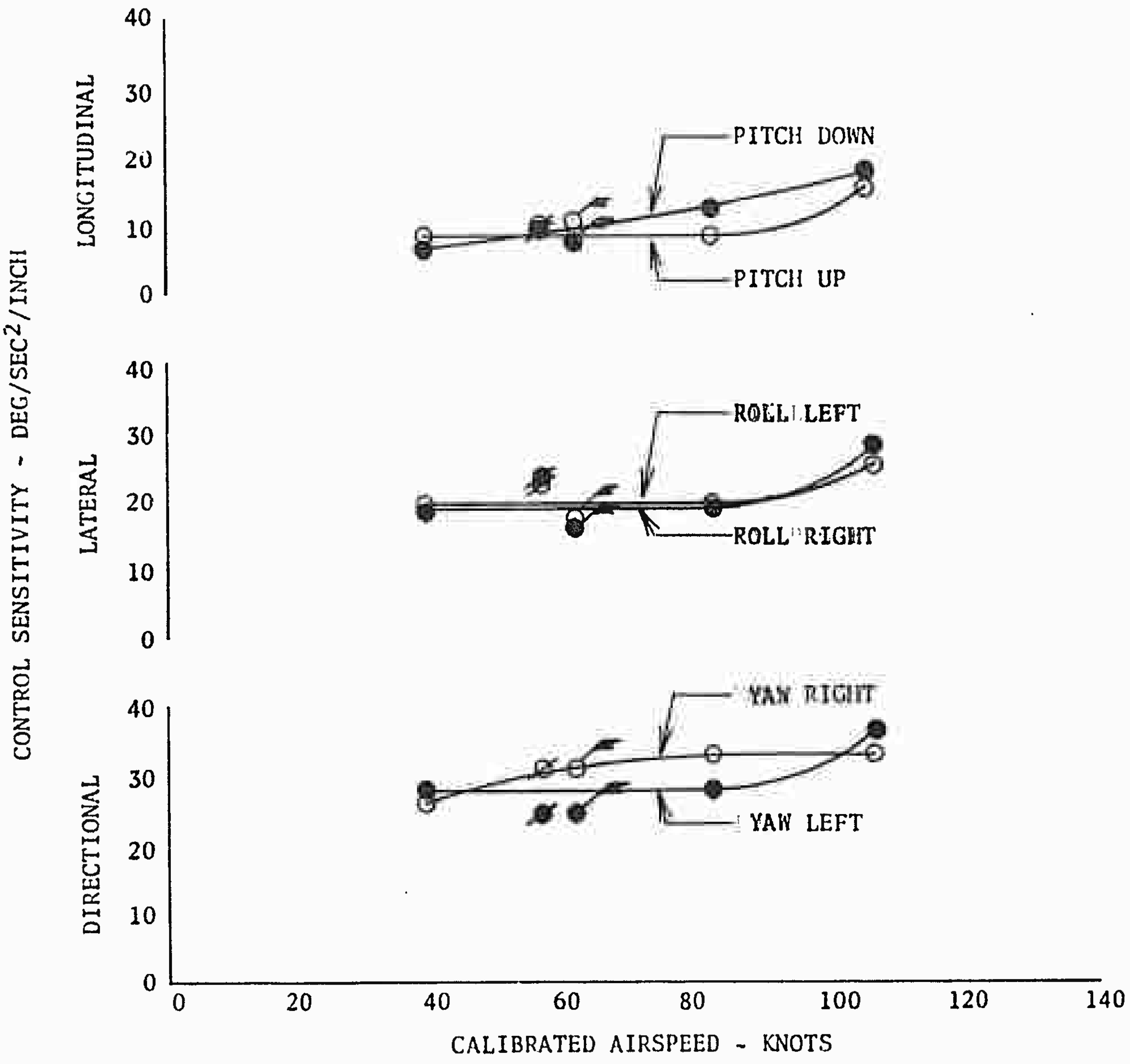


FIGURE NO.52
SUMMARY OF CONTROL RESPONSE
UH-1B/540 USA S/N 64-18105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
◇	9386	5000	126.5	324	XM-21/M-5
◇ (slashed, flagged)	9395	5000	126.5	324	XM-21/M-5
◇ (flagged)	9395	5000	126.5	324	XM-21/M-5
◁	9290	5000	126.4	324	XM-16/M-5
◁ (slashed, flagged)	9230	5000	126.3	324	XM-16/M-5
◁ (flagged)	9230	5000	126.3	324	XM-16/M-5

NOTES:
1. POINTS OBTAINED FROM FIGURE NO. 59 THROUGH 61
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. FLAGGED SYMBOL DENOTES AUTOROTATION.
4. SLASHED SYMBOL DENOTES CLIMB.
5. CONTROL RESPONSE VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

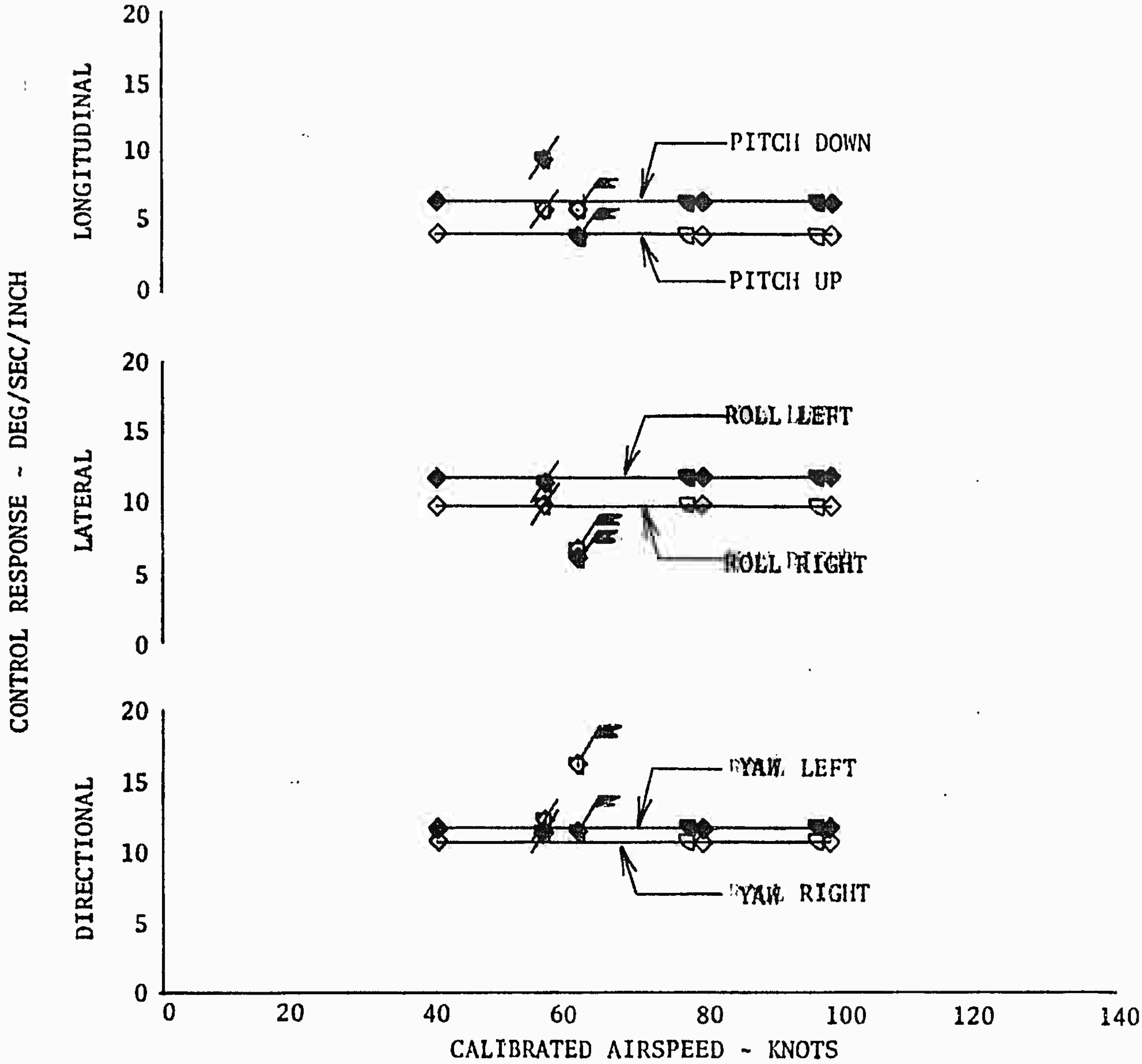


FIGURE NO. 53
SUMMARY OF CONTROL SENSITIVITY
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. STATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
◇	9386	5000	126.5	324	XM-21/M-5
◇ (slashed, flagged)	9395	5000	126.5	324	XM-21/M-5
◇ (flagged)	9395	5000	126.5	324	XM-21/M-5
◁	9290	5000	126.4	324	XM-16/M-5
◁ (slashed, flagged)	9230	5000	126.3	324	XM-16/M-5
◁ (flagged)	9230	5000	126.3	324	XM-16/M-5

NOTES:
1. POINTS OBTAINED FROM FIGURE NO. 59 THROUGH 61
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. FLAGGED SYMBOL DENOTES AUTOROTATION.
4. SLASHED SYMBOL DENOTES CLIMB.
5. CONTROL SENSITIVITY VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

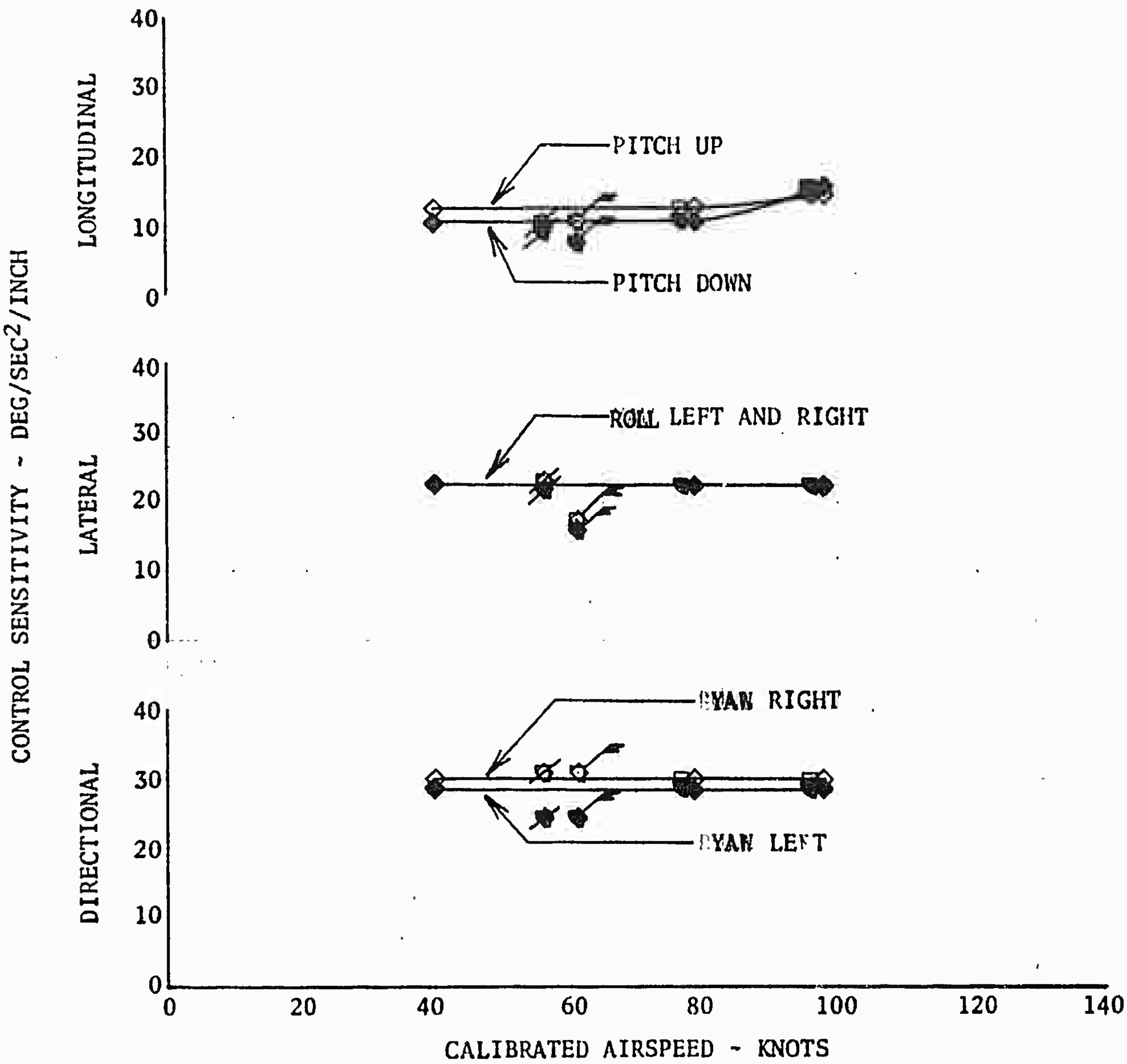


FIGURE NO. 54
SUMMARY OF CONTROL RESPONSE
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
△	7927	5000	126.6	324	XM-3/M-5

NOTES:
1. POINTS OBTAINED FROM FIGURES 59 THROUGH 61
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. CONTROL RESPONSE VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

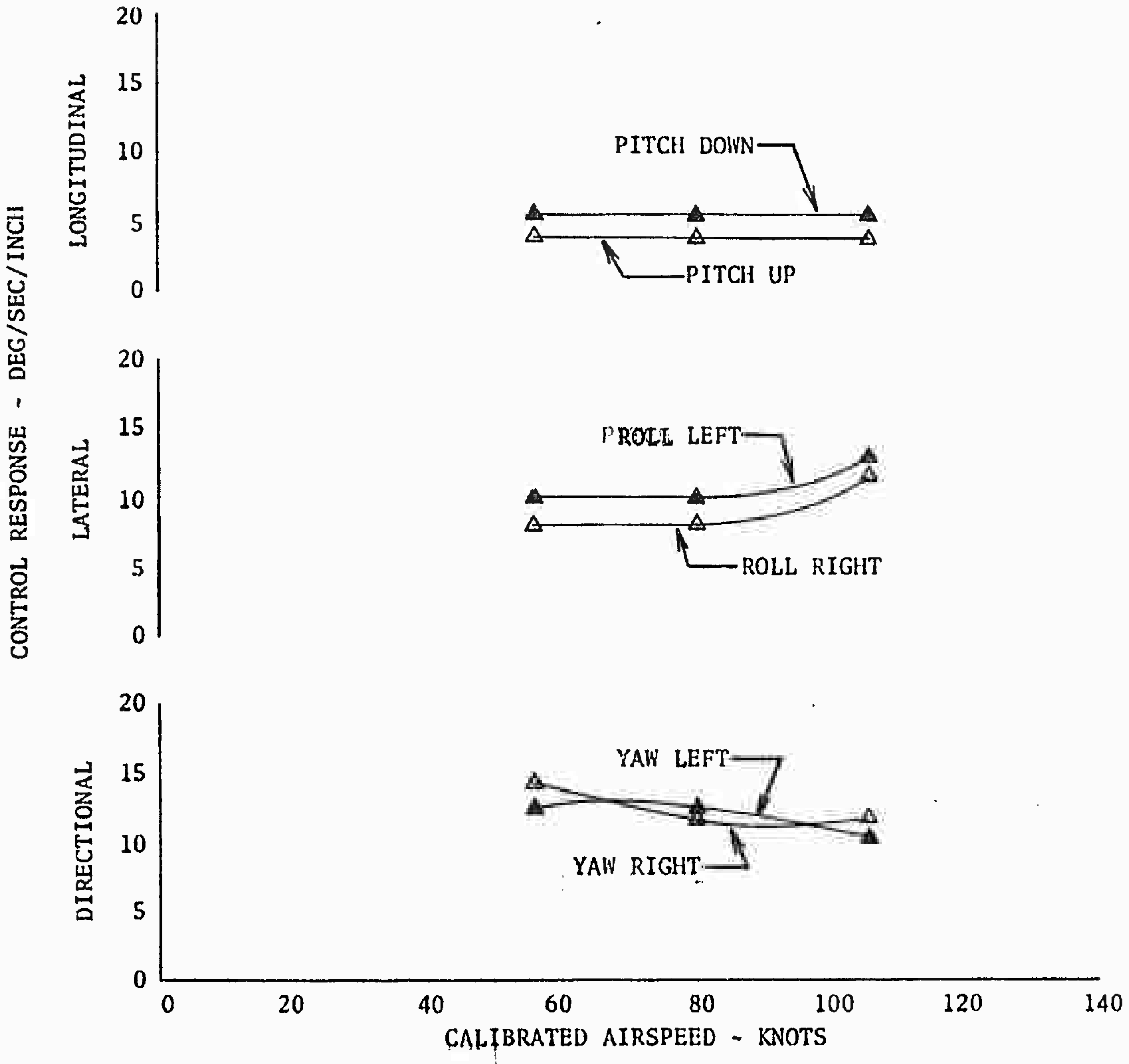


FIGURE NO. 55
SUMMARY OF CONTROL SENSITIVITY
UH-1B/540 USA S/N 64-14105

SYM	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
Δ	7927	5000	126.6	324	XM-3/M-5

NOTES:
1. POINTS OBTAINED FROM FIGURES 62 THROUGH 64
2. SHADED SYMBOL DENOTES LEFT ROLL, LEFT YAW, OR PITCH DOWN.
3. CONTROL SENSITIVITY VALUES OBTAINED AT A CONTROL DISPLACEMENT OF 1 INCH FROM TRIM.

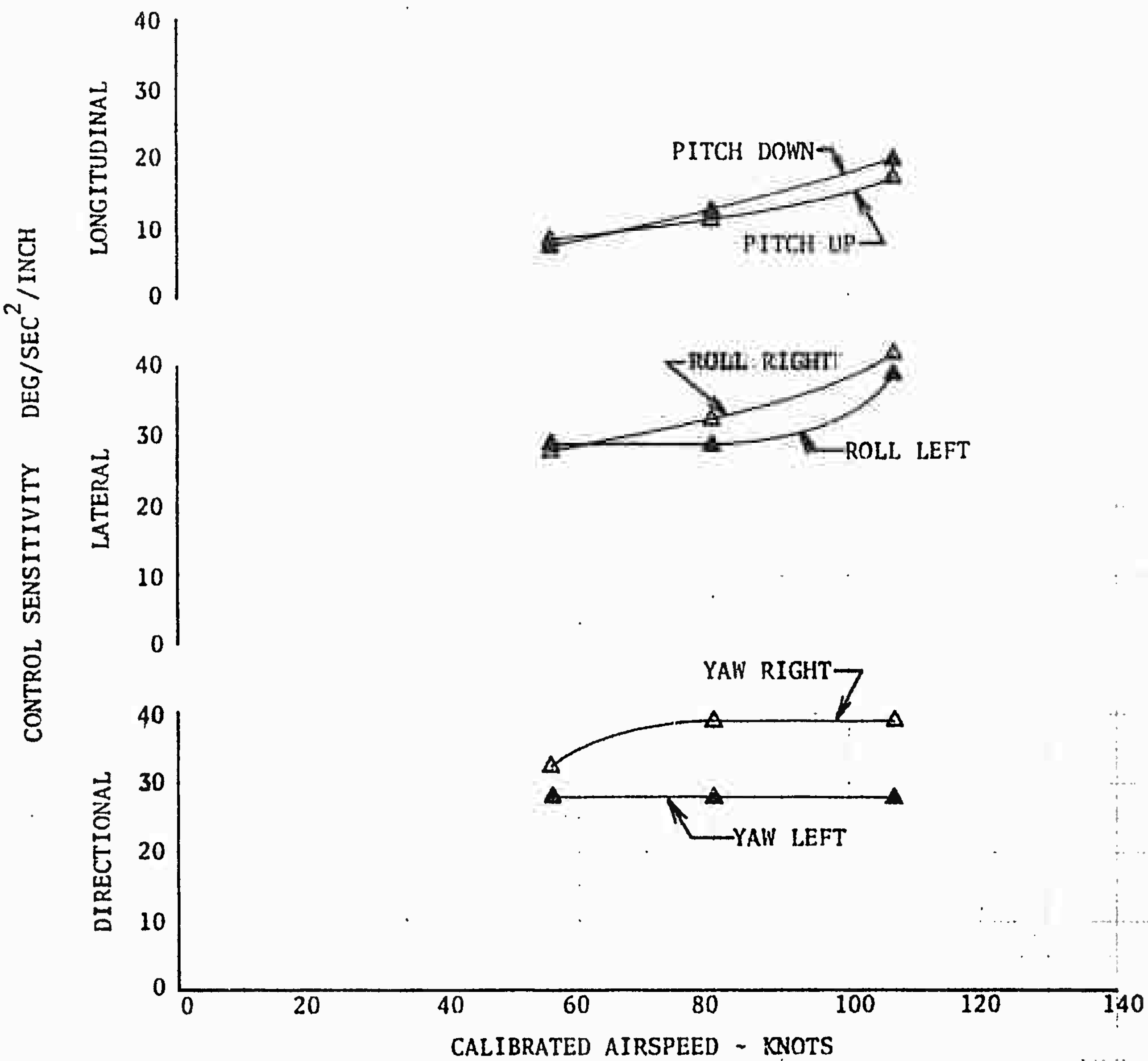


FIGURE NO. 56
LONGITUDINAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	39	7822	5000	126.4	324
□	82	7686	5000	126.2	324
△	106	7573	5000	126.1	324

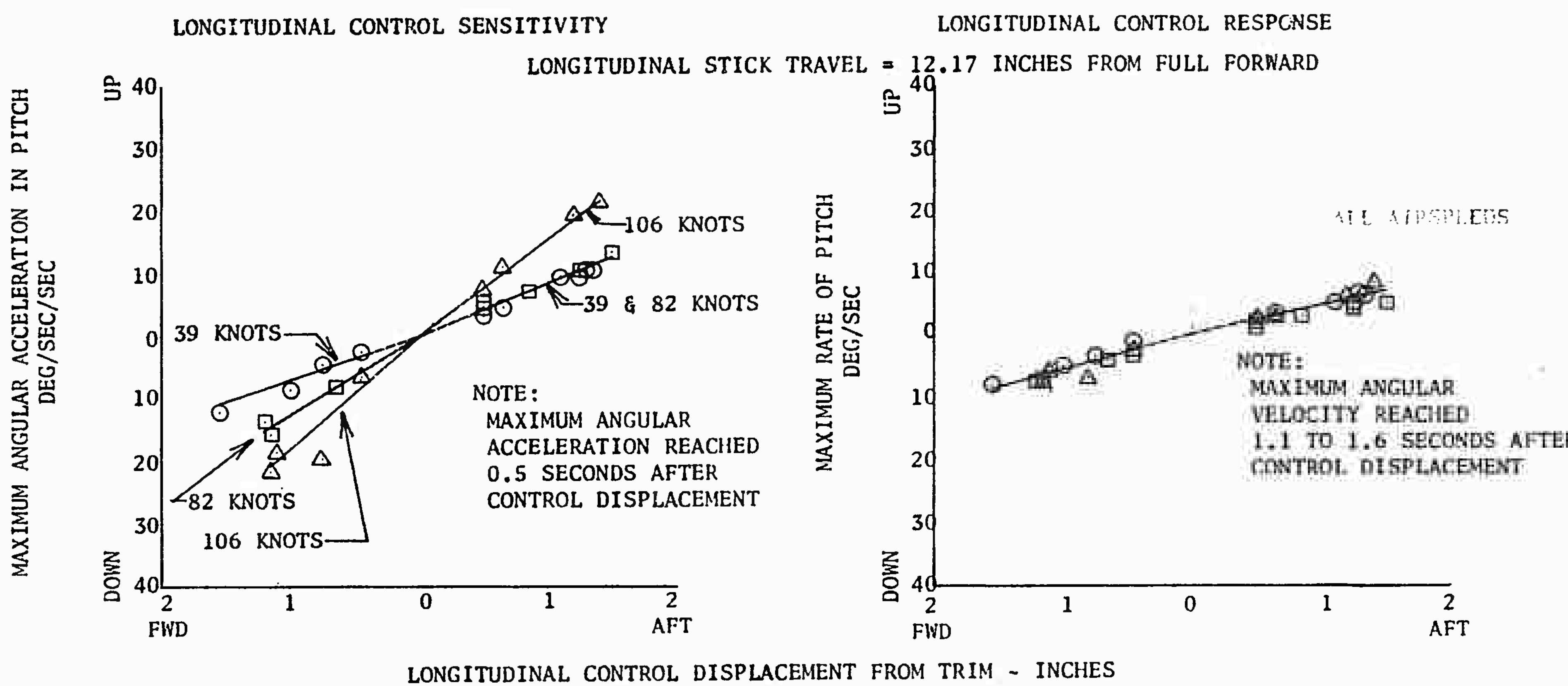


FIGURE NO. 57
DIRECTIONAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	39	7920	5000	126.6	324
□	82	7719	5000	126.3	324
△	106	7561	5000	126.0	324

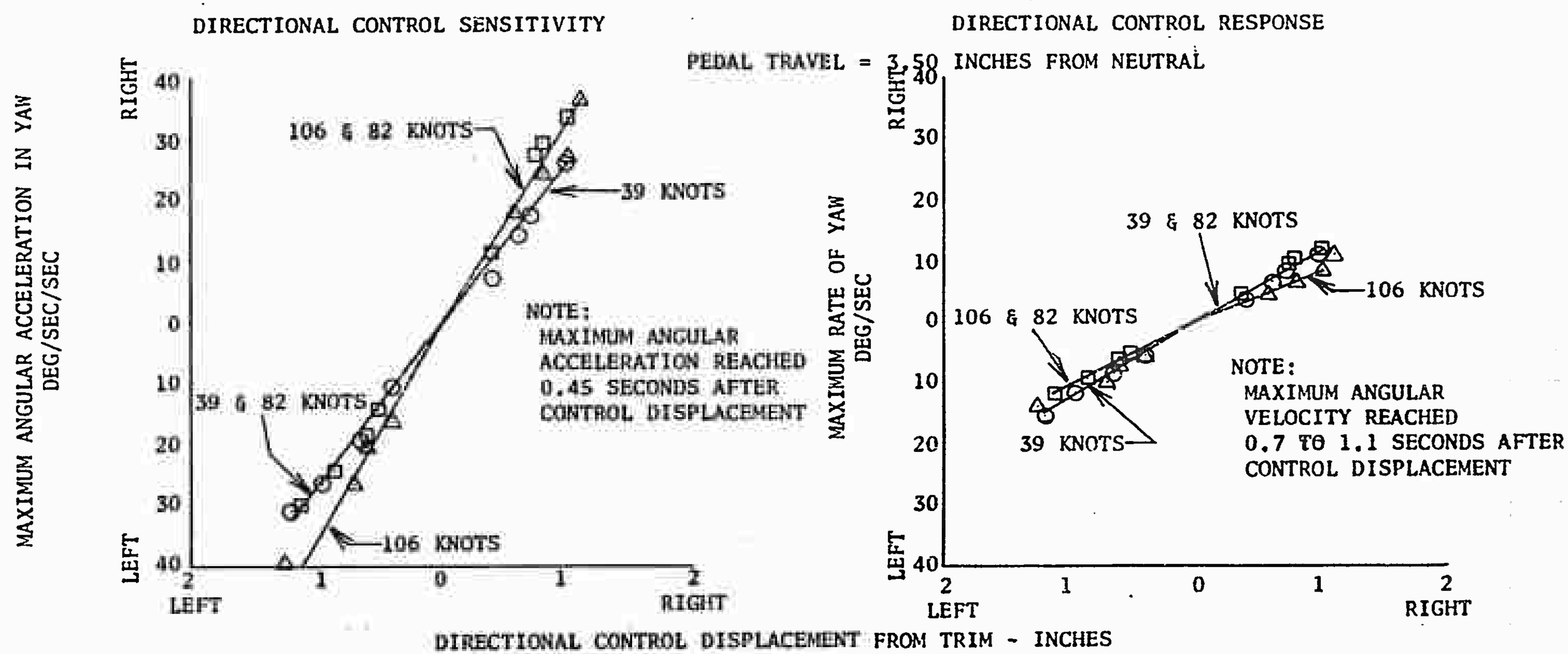


FIGURE NO. 58
LATERAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	39	7920	5000	126.6	324
□	82	7719	5000	126.3	324
△	106	7561	5000	126.0	324

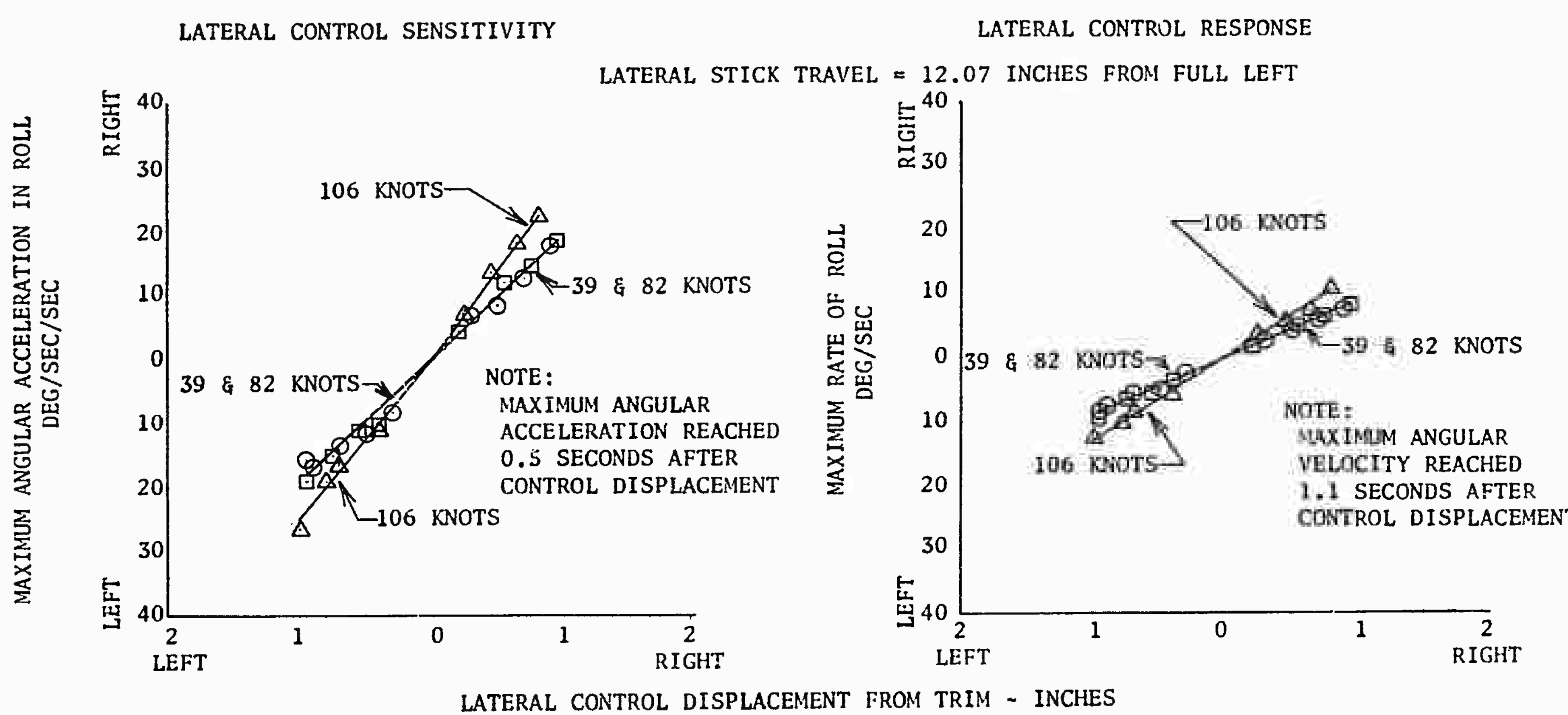


FIGURE NO. 59
LONGITUDINAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	40	9527	5000	126.8	324
◁	77	9437	5000	126.6	324
□	79	9550	5000	126.8	324
⬭	96	9264	5000	126.3	324
△	98	9338	5000	126.5	324

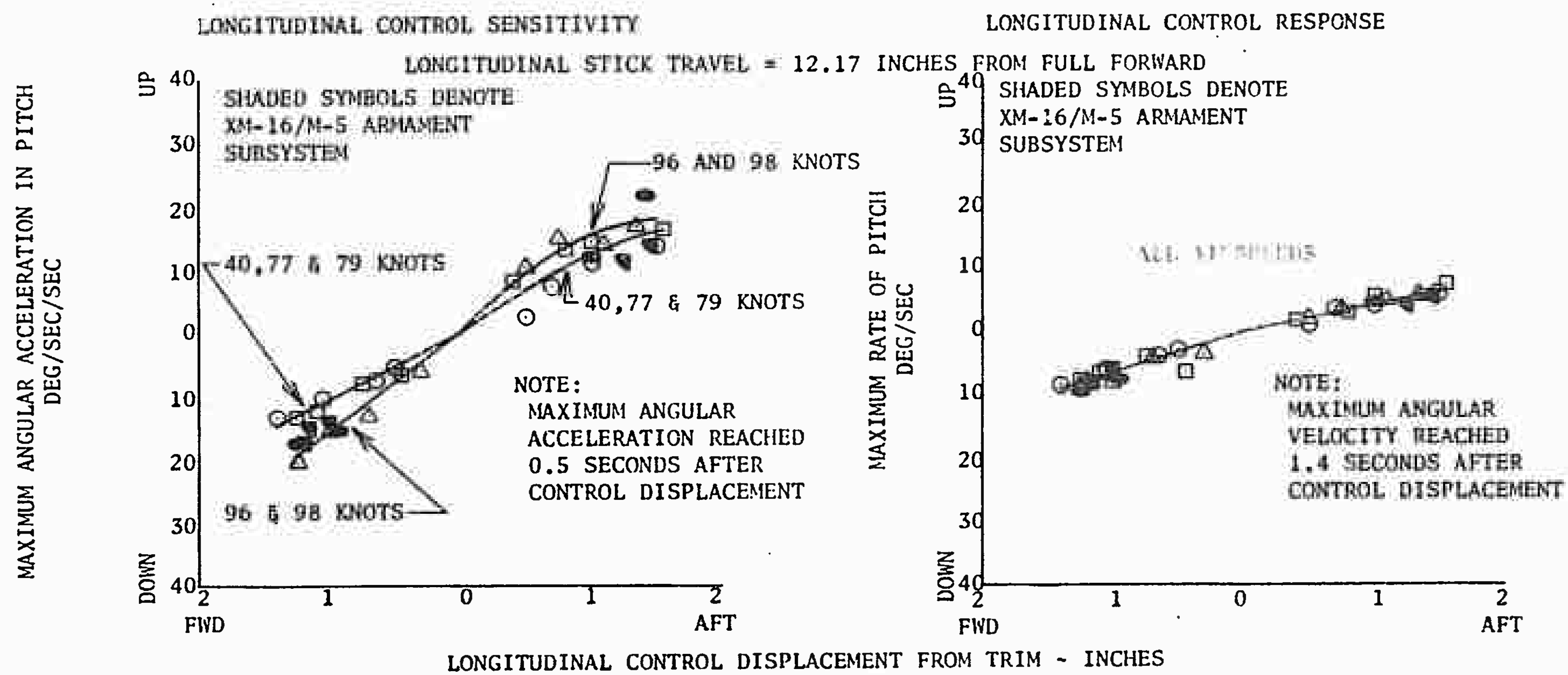


FIGURE NO. 60
LATERAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	40	9438	5000	126.4	324
◁	77	9384	5000	126.5	324
□	79	9483	5000	126.7	324
0	96	9196	5000	126.3	324
△	98	9238	5000	126.4	324

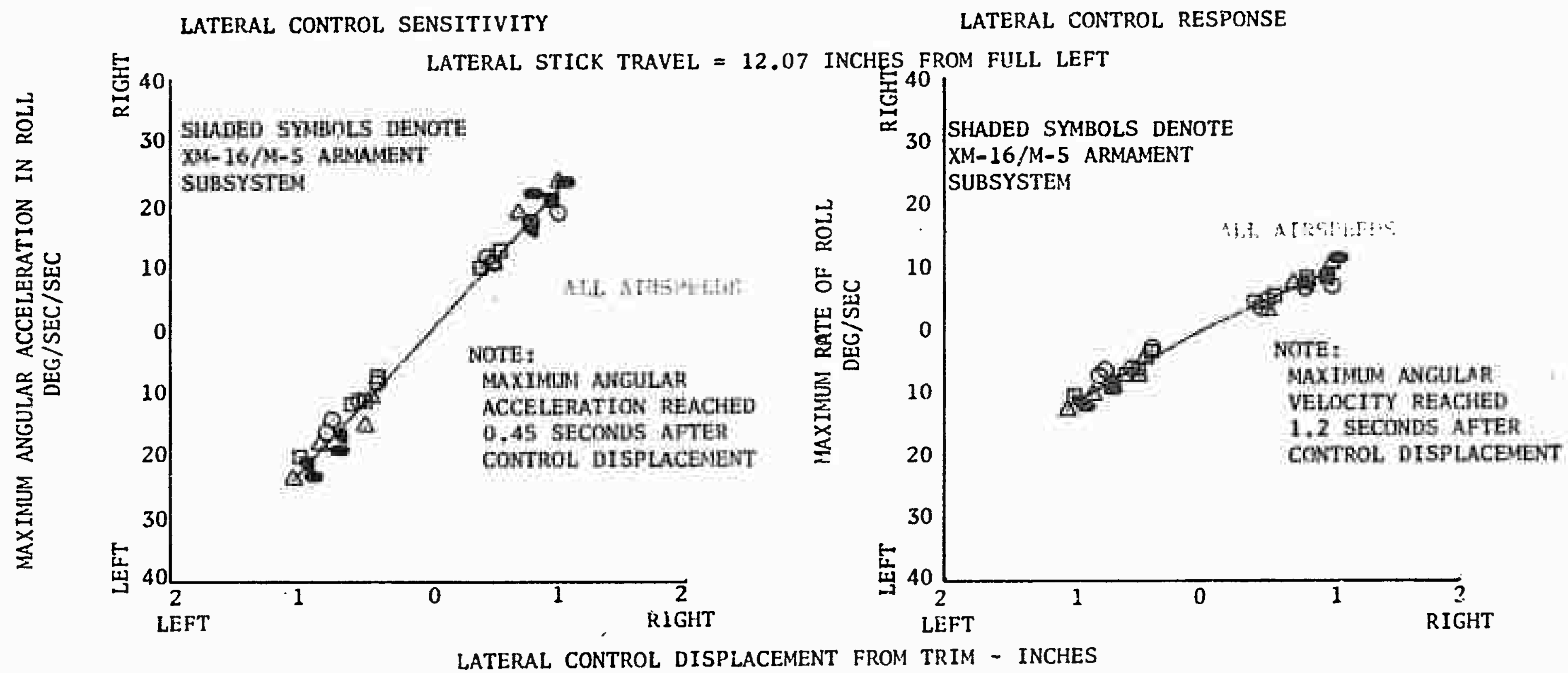


FIGURE NO. 61
DIRECTIONAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
XM-21/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	40	9238	5000	126.4	324
◁	77	9338	5000	126.4	324
□	79	9400	5000	126.6	324
○	96	9137	5000	126.2	324
△	98	9319	5000	126.5	324

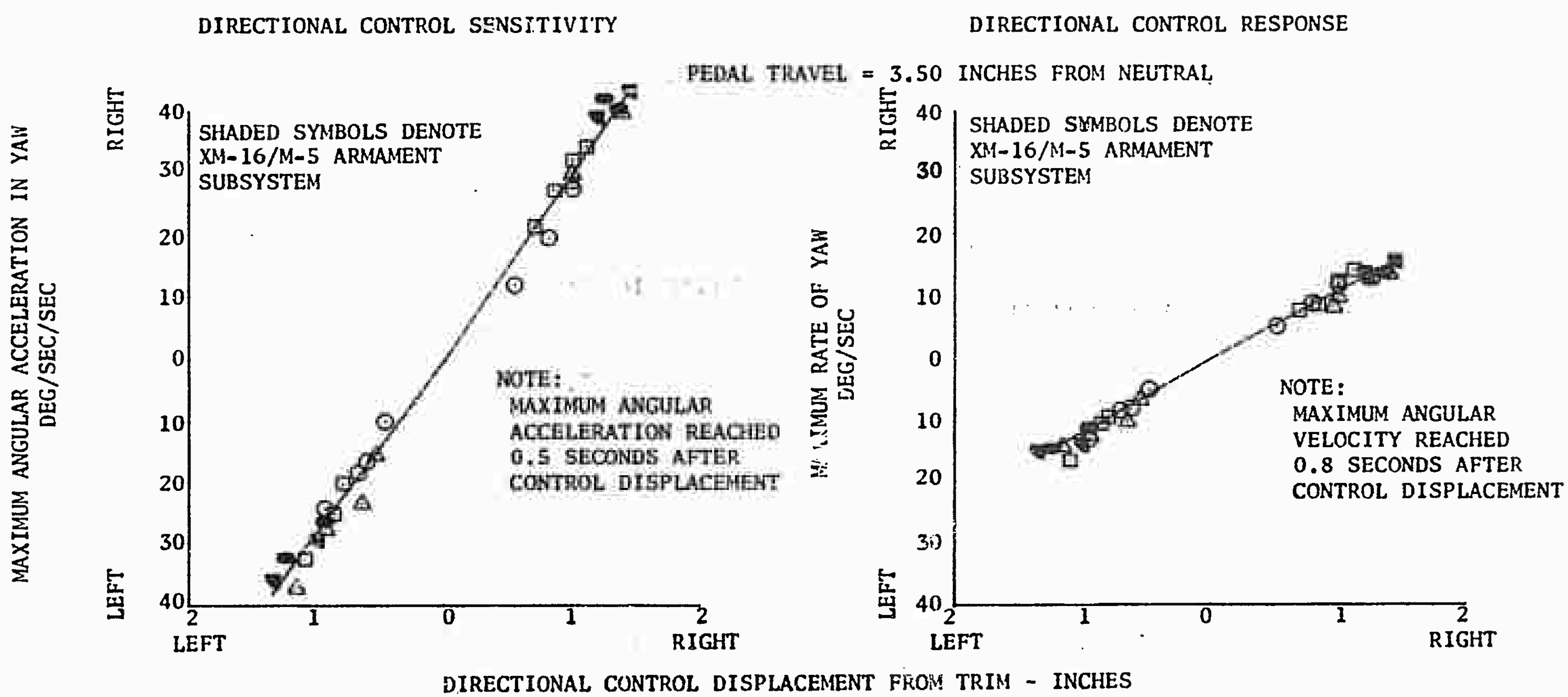


FIGURE NO. 62
LONGITUDINAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
M-3/M-5 ARMAMENT SUBSYSTEM
LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	CROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	56	7820	5000	126.5	324
□	80	7750	5000	126.4	324
△	107	7740	5000	126.4	324

LONGITUDINAL CONTROL SENSITIVITY

LONGITUDINAL CONTROL RESPONSE

LONGITUDINAL STICK TRAVEL = 12.17 INCHES FROM FULL FORWARD

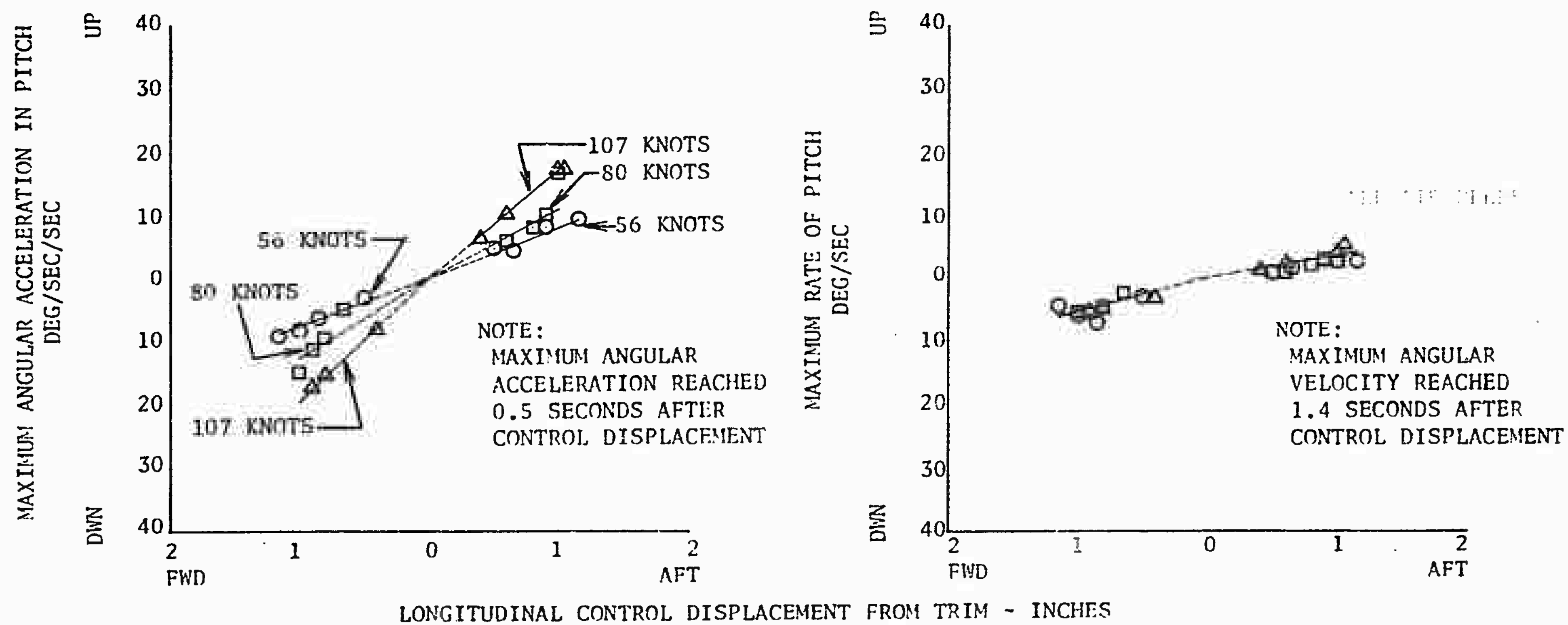


FIGURE NO. 63
LATERAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
M-3/M-5 ARMAMENT SUBSYSTEM
LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	56	7950	5000	126.7	324
□	80	7910	5000	126.6	324
△	107	7920	5000	126.6	324

LATERAL CONTROL SENSITIVITY

LATERAL CONTROL RESPONSE

LATERAL STICK TRAVEL = 12.07 INCHES FROM FULL LEFT

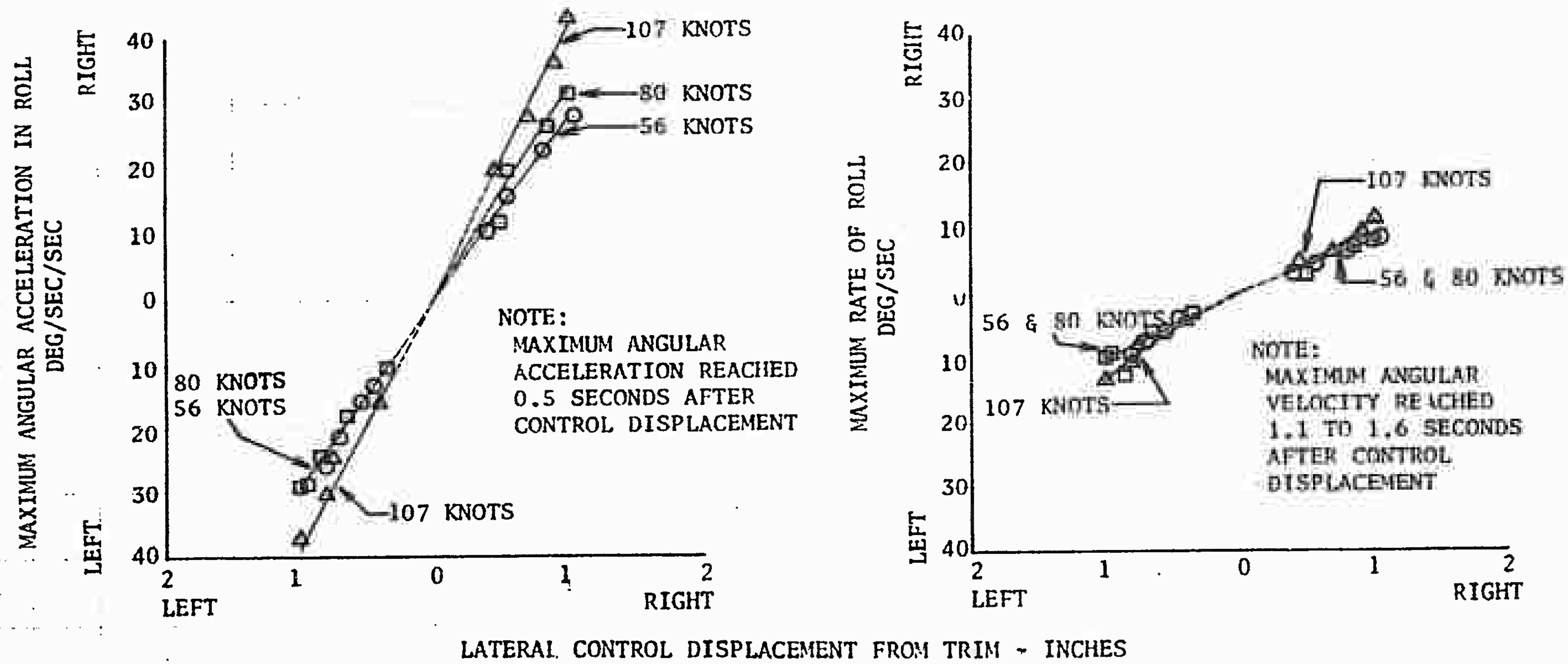
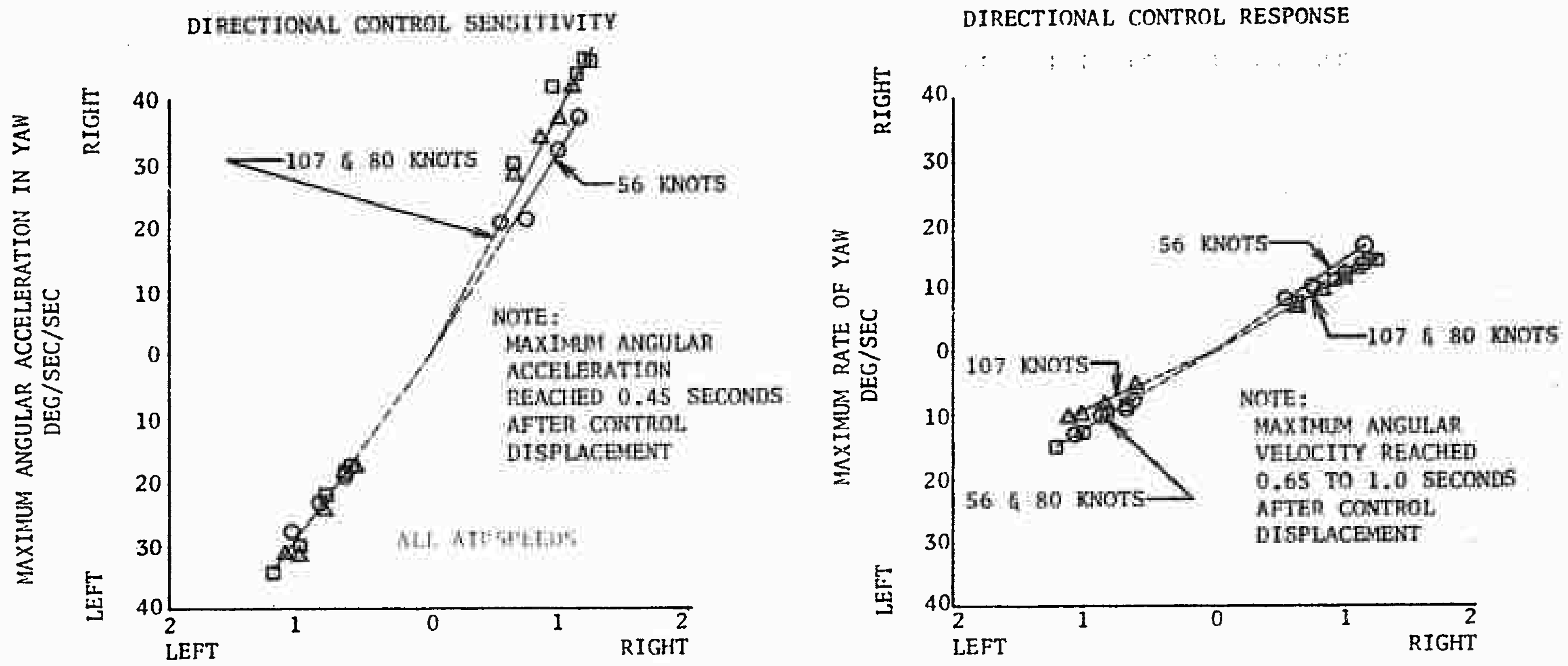


FIGURE NO. 64
DIRECTIONAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
M-3/M-5 ARMAMENT SUBSYSTEM

LEVEL FLIGHT

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM
○	56	7645	5000	126.3	324
□	80	7595	5000	126.2	324
△	107	7550	5000	126.2	324

PEDAL TRAVEL = 3.50 INCHES FROM NEUTRAL



DIRECTIONAL CONTROL DISPLACEMENT FROM TRIM - INCHES

FIGURE NO. 65
LONGITUDINAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
CLIMB AND AUTOROTATION

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
△	56	7688	5000	126.2	324	XM-21/M-5
⬠	61	7688	5000	126.2	324	XM-21/M-5
◇	56	9451	5000	126.6	324	XM-21/M-5
▽	61	9451	5000	126.6	324	XM-21/M-5
○	56	9465	5000	126.6	324	XM-16/M-5
□	61	9465	5000	126.6	324	XM-16/M-5

LONGITUDINAL CONTROL SENSITIVITY

LONGITUDINAL CONTROL RESPONSE

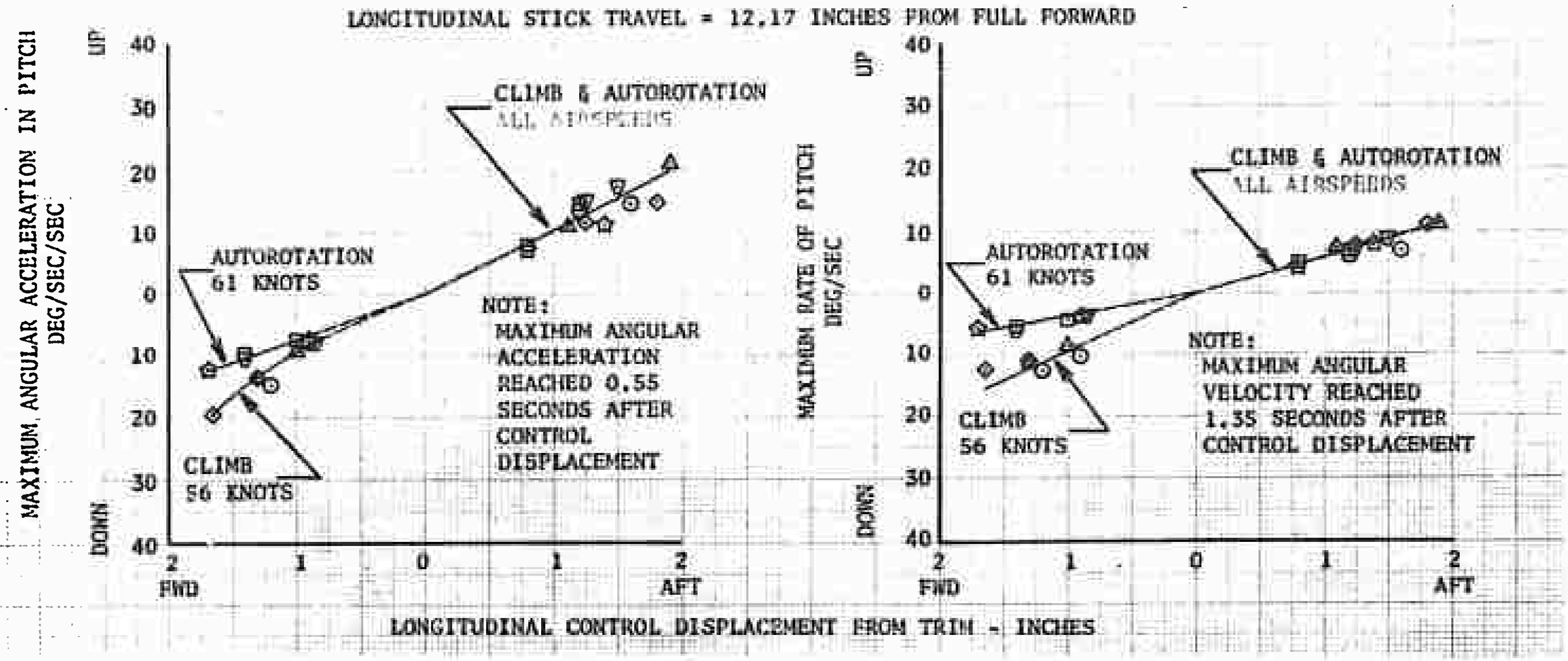


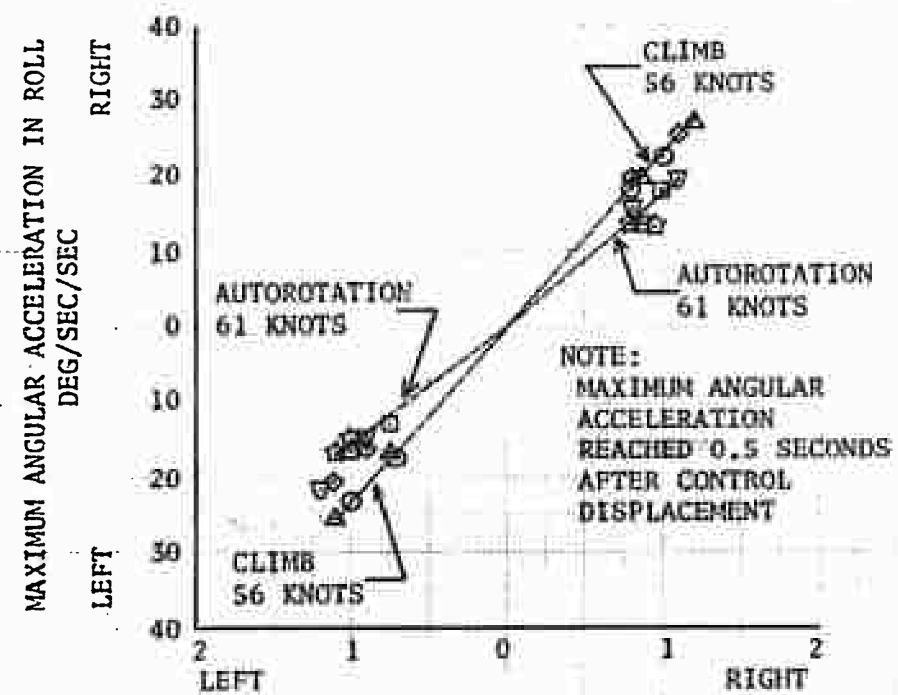
FIGURE NO.66
LATERAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
CLIMB AND AUTOROTATION

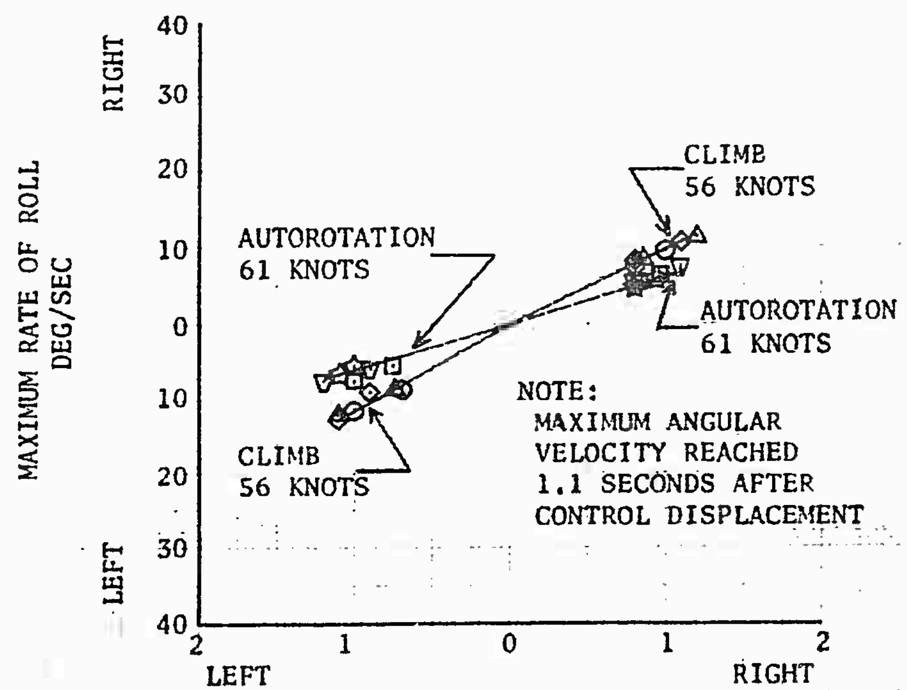
SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
△	56	7722	5000	126.3	324	XM-21/M-5
✩	61	7722	5000	126.3	324	XM-21/M-5
◇	56	9262	5000	126.4	324	XM-21/M-5
▽	61	9262	5000	126.4	324	XM-21/M-5
○	56	9305	5000	126.4	324	XM-16/M-5
□	61	9305	5000	126.4	324	XM-16/M-5

LATERAL CONTROL SENSITIVITY

LATERAL CONTROL RESPONSE

LATERAL STICK TRAVEL = 12.07 INCHES FROM FULL LEFT





LATERAL CONTROL DISPLACEMENT FROM TRIM ~ INCHES

FIGURE NO. 67
DIRECTIONAL CONTROLLABILITY
UH-1B/540 USA S/N 64-14105
CLIMB AND AUTOROTATION

SYM	CALIBRATED AIRSPEED KTS	GROSS WEIGHT LBS	DENSITY ALTITUDE FT	C.G. LOCATION IN	ROTOR SPEED RPM	ARMAMENT SUBSYSTEM
△	56	7878	5000	126.5	324	XM-21/M-5
✩	61	7878	5000	126.5	324	XM-21/M-5
◇	56	9395	5000	126.5	324	XM-21/M-5
▽	61	9395	5000	126.5	324	XM-21/M-5
○	56	9230	5000	126.3	324	XM-16/M-5
□	61	9230	5000	126.3	324	XM-16/M-5

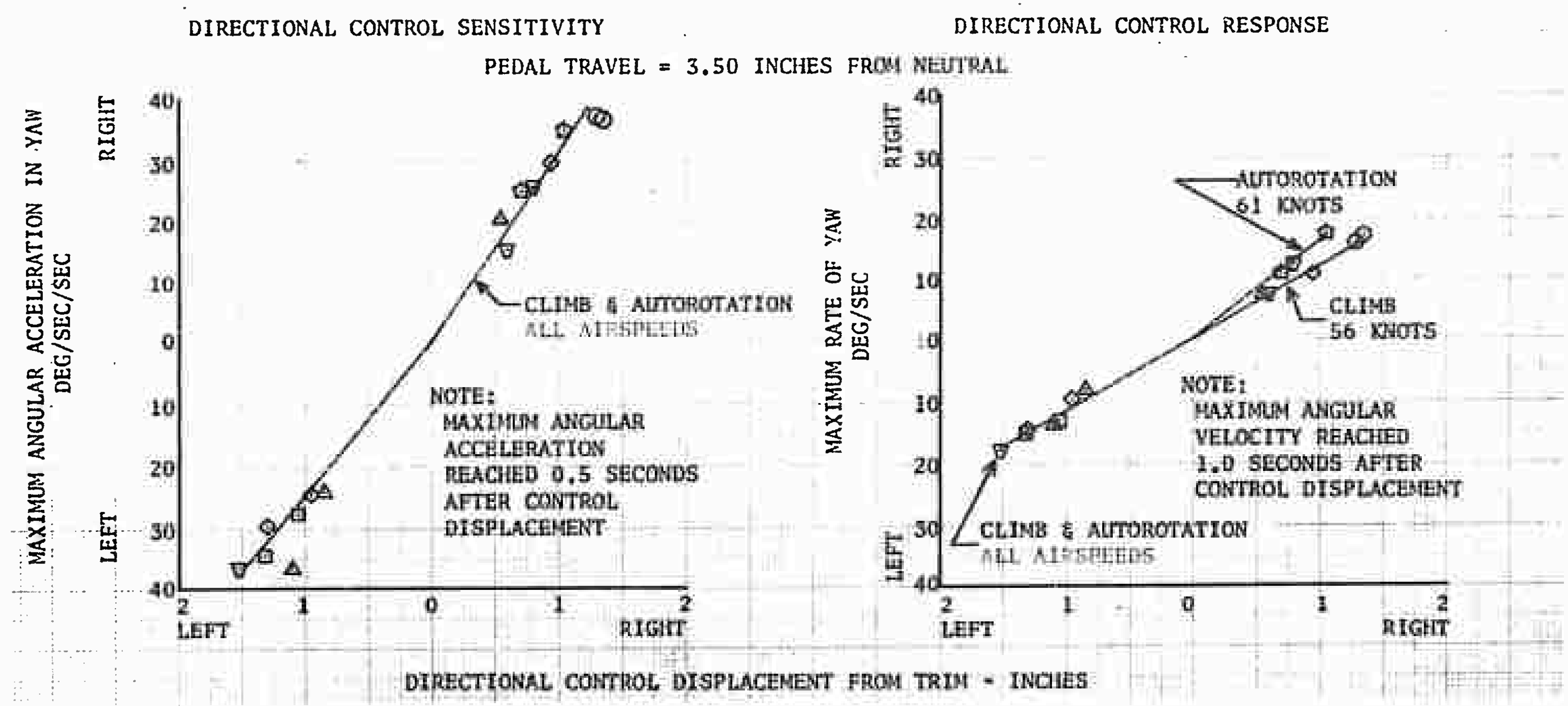


FIGURE NO. 68
AIRSPEED CALIBRATION
UH-1B/540 USA S/N 64-14105
LEVEL FLIGHT
SHIP SYSTEM
TRAILING BOMB METHOD

GROSS WEIGHT = 7700 LBS
C.G. LOCATION = 127.2 (FWD)
ROTOR SPEED = 324 RPM

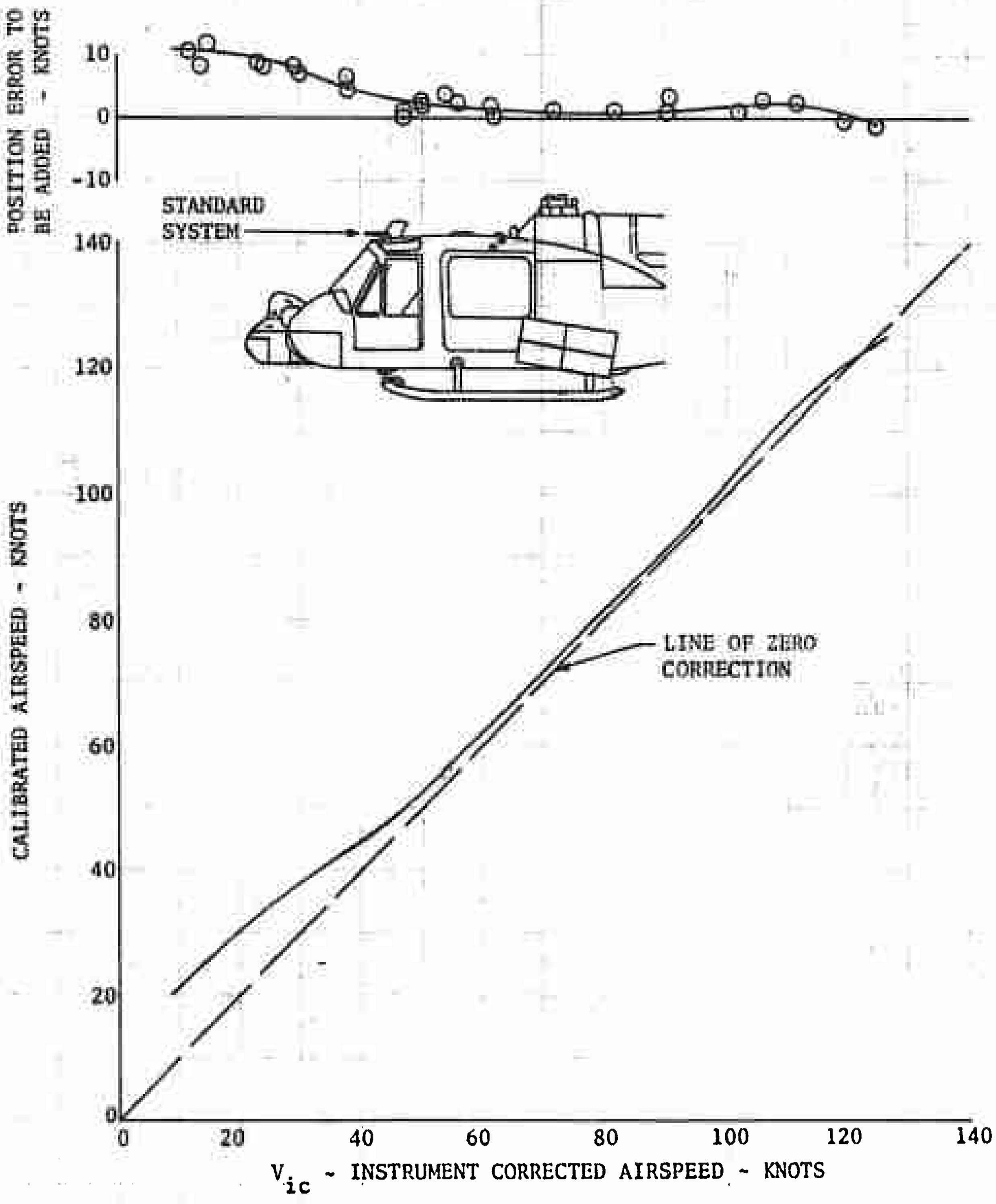


FIGURE NO. 69
JETTISON TEST

UH-1B/540 XM-3/M-5 ARMAMENT SUBSYSTEM S/N 64-14105

LEVEL FLIGHT

ARMAMENT CONFIGURATION
TWO EMPTY ROCKET LAUNCHERS
ROCKET LAUNCHER ANGLE = 6 DEGREES NOSE UP

ANGLE OF ATTACK = 7.0 DEG. DOWN AVG. WEIGHT = 8100 LBS
ANGLE OF SIDESLIP = 17.0 DEG LEFT AVG. C.G. LOC. = 126.7 (FWD)
ROTOR SPEED = 324 RPM DEN. ALTITUDE = 3390 FEET

CALIBRATED AIRSPEED = 80 KNOTS

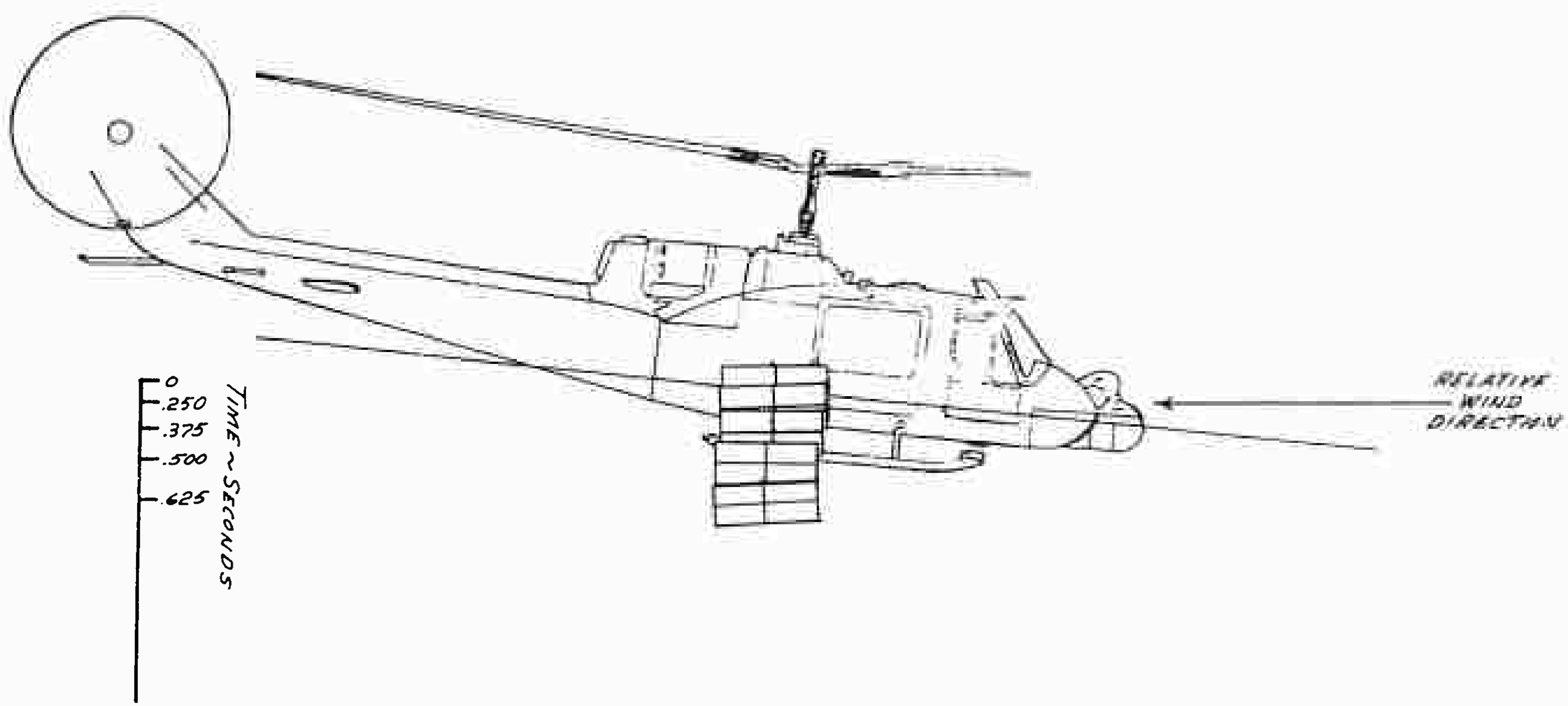


FIGURE NO. 70
JETTISON TEST
UH-1B/540 XM-3/M-5 ARMAMENT SUBSYSTEM USA S/N 64-14105
POWERED DESCENT
ARMAMENT CONFIGURATION
TWO EMPTY ROCKET LAUNCHERS
ROCKET LAUNCHER ANGLE 6 DEGREES NOSE UP

ANGLE OF ATTACK = 2.5 DEG DOWN
ANGLE OF SIDESLIP = 0 DEG
ROTOR SPEED = 324 RPM
AVG WEIGHT = 8100 LBS
AVG. C.G. LOC. = 126.7 (FWD)
DEN. ALTITUDE = 3910 FEET
CALIBRATED AIRSPEED = 130 KNOTS

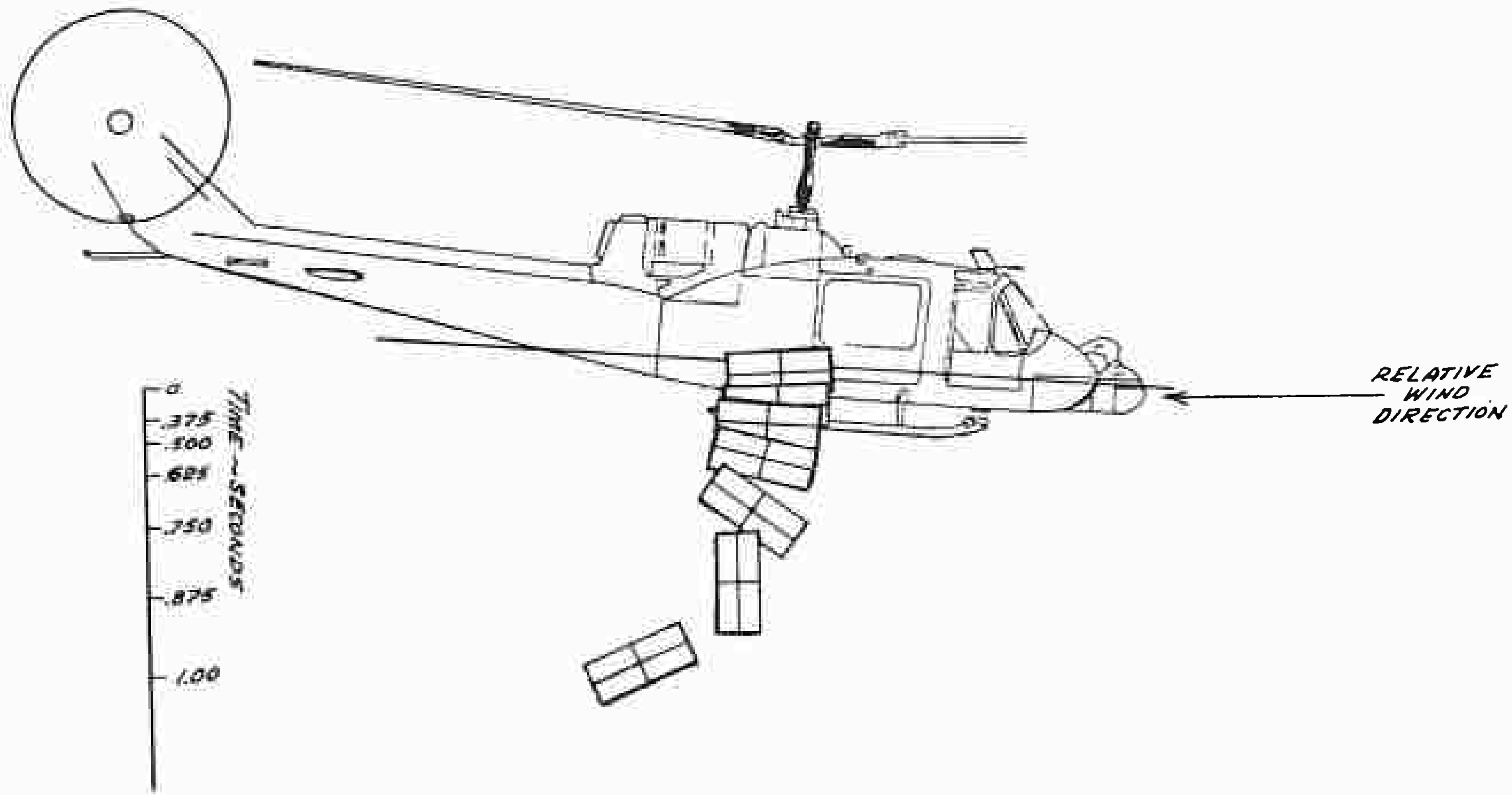
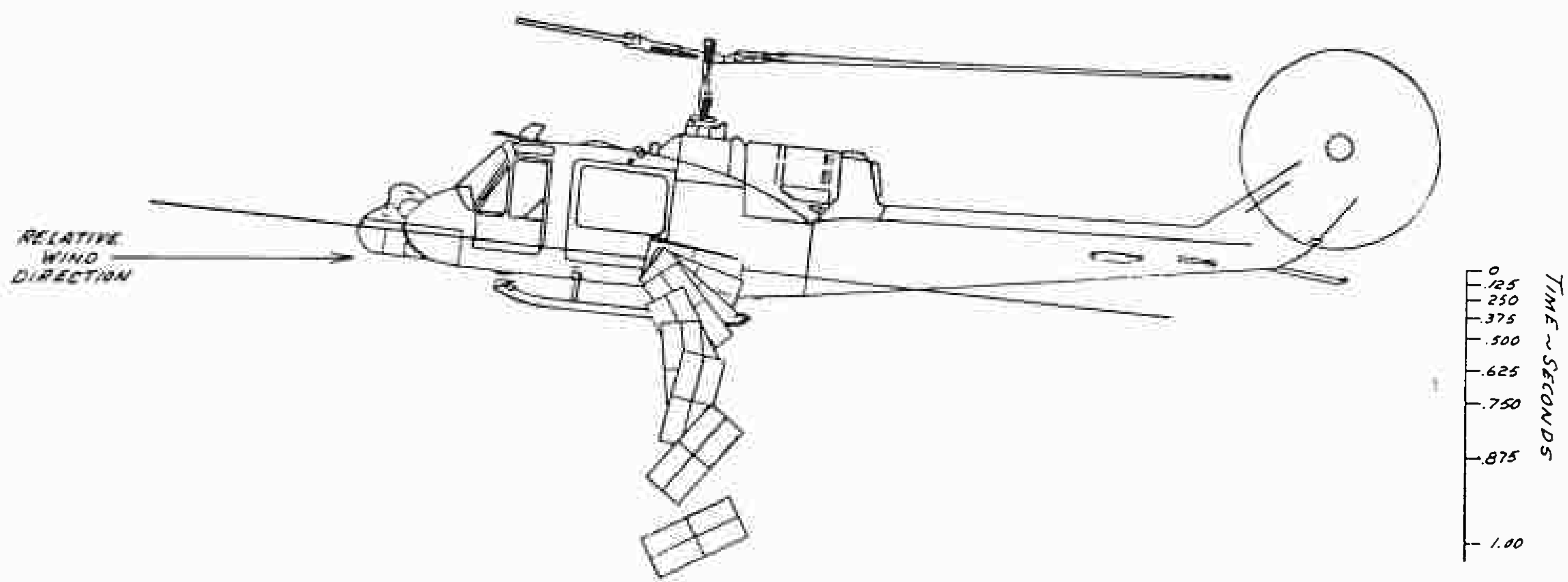


FIGURE NO. 71
JETTISON TEST
UH-1B/540 XM-3/M-5 ARMAMENT SUBSYSTEM USA S/N 64-14105
AUTOROTATION
ARMAMENT CONFIGURATION
TWO EMPTY ROCKET LAUNCHERS
ROCKET LAUNCHER ANGLE 6 DEGREES NOSE UP

ANGLE OF ATTACK = 10.5 DEG. UP
ANGLE OF SIDESLIP = 4.0 DEG. RIGHT
ROTOR SPEED = 324 RPM

AVG. WEIGHT = 8100
AVG. C.G. LOC. = 126.7 (FWD)
DEN. ALTITUDE = 3470 FT.

CALIBRATED AIRSPEED = 60 KNOTS



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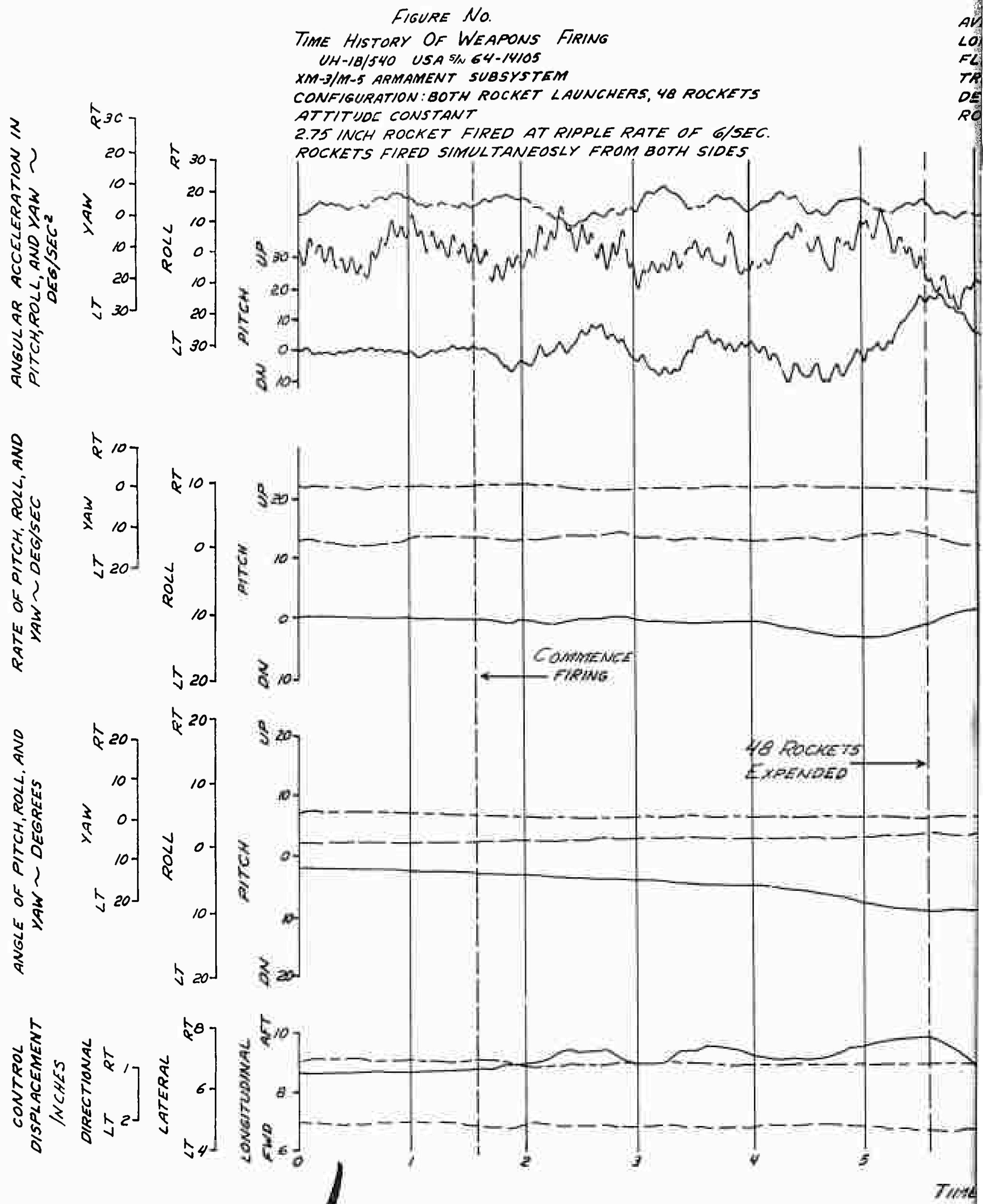

FIGURE NO.
TIME HISTORY OF WEAPONS FIRING
UH-1B/540 USA S/N 64-14105
XM-3/M-5 ARMAMENT SUBSYSTEM
CONFIGURATION: BOTH ROCKET LAUNCHERS, 48 ROCKETS
ATTITUDE CONSTANT
2.75 INCH ROCKET FIRED AT RIPPLE RATE OF 6/SEC.
ROCKETS FIRED SIMULTANEOSLY FROM BOTH SIDES

AVERAGE GROSS WEIGHT: 9000 LBS.
LONGITUDINAL C.G. LOCATION: 127.4 IN (FWD)
FLIGHT CONDITIONS: HOVER
TRIM C.A.S.: ZERO
DENSITY ALTITUDE: 2180 FT.
ROTOR SPEED: 324 RPM

——— LONGITUDINAL STICK
— — — LATERAL STICK
— - — DIRECTIONAL

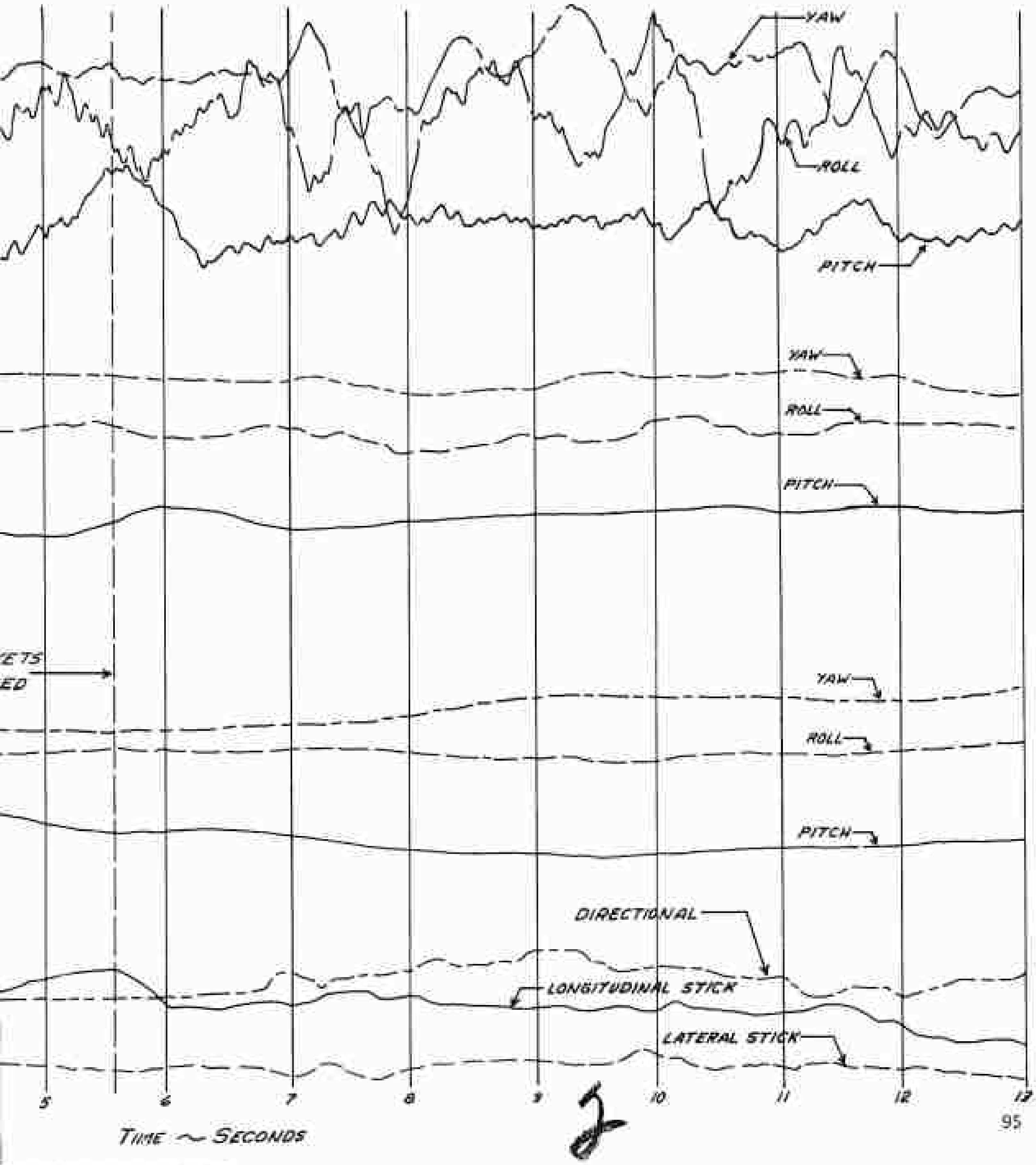

FIGURE NO. 73
ENGINE TORQUEMETER CALIBRATION
UH-1B/540 USA S/N 64-14105

NOTE: DATA TAKEN FROM LYCOMING "GREEN RUN"
SHEETS DATED 19 MARCH 1965
ENGINE MODEL T53-L-11
ENGINE SERIAL NO. LE 10382

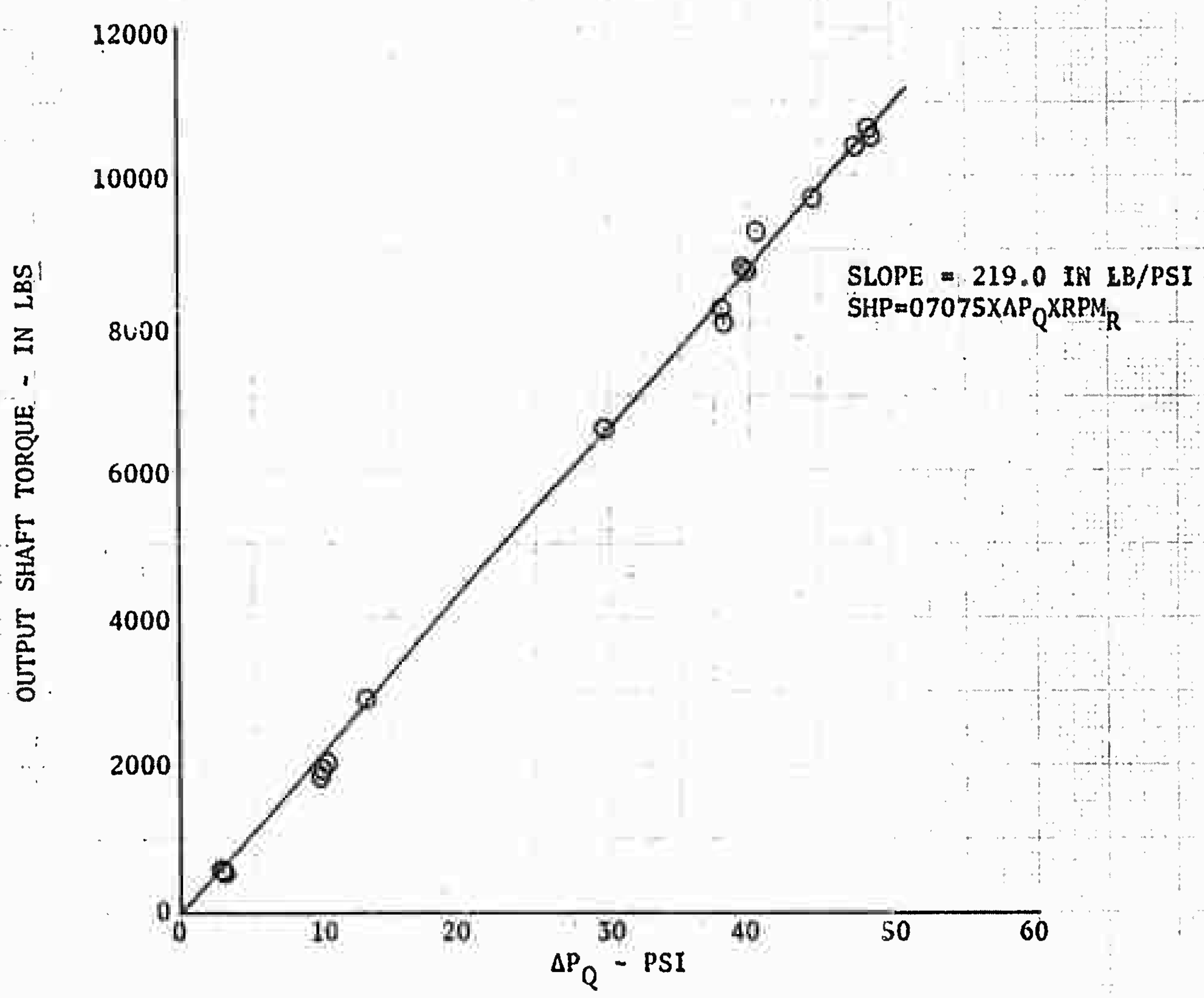

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FIGURE NO. 74
MILITARY POWER AVAILABLE
STANDARD DAY
MODEL SPEC. T53-L-11 ENGINE
6600 RPM
2°C INLET TEMPERATURE RISE

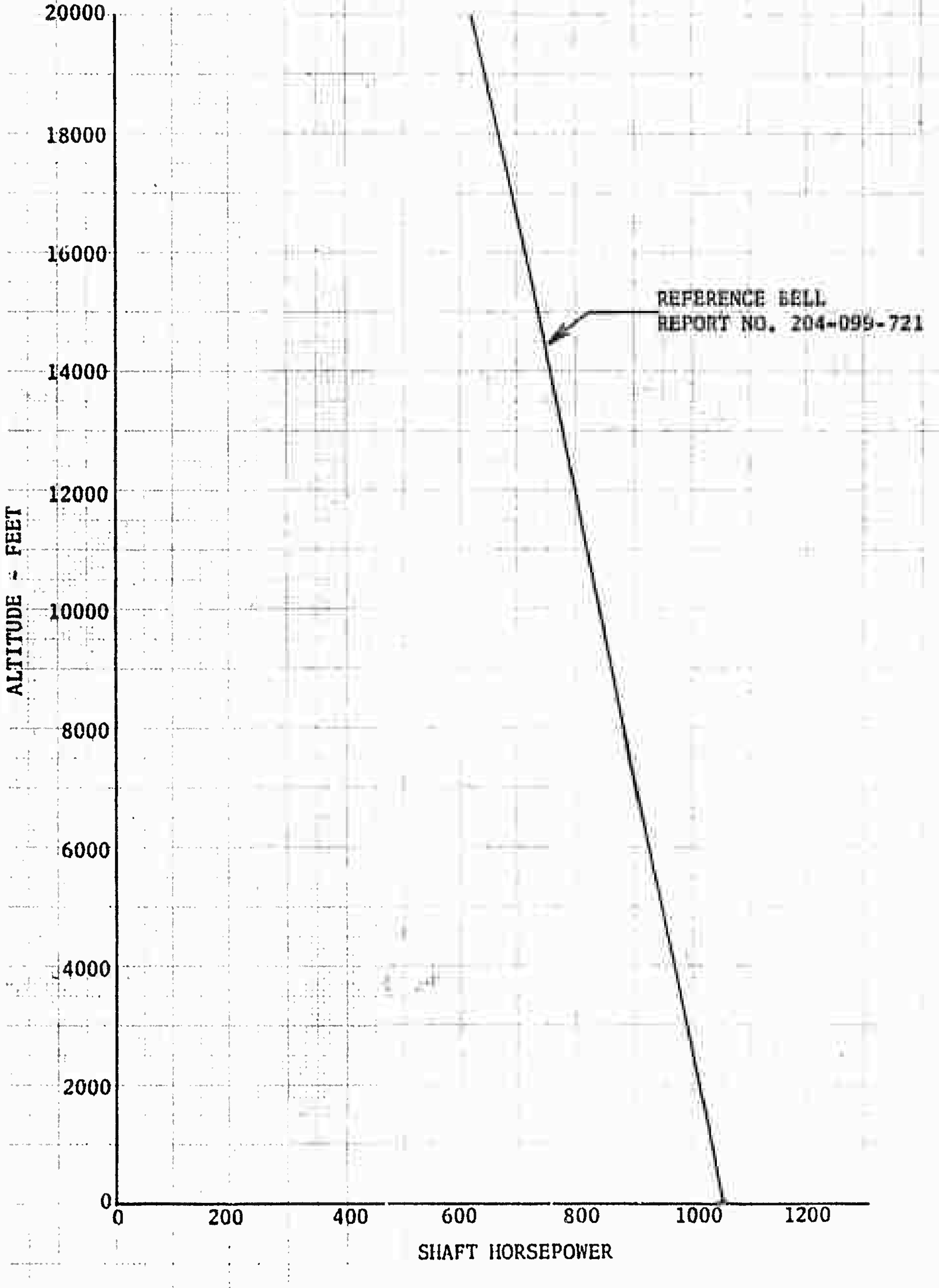


FIGURE NO. 75
NORMAL RATED SHAFT HORSEPOWER AVAILABLE
UH-1B/540
T53-L-11

NOTES:

1. SHAFT HORSEPOWER AVAILABLE BASED ON LYCOMING T53-L-11 ENGINE MODEL SPECIFICATION.
2. COMPRESSOR INLET TEMPERATURE RISE = +2°C
3. COMPRESSOR INLET PRESSURE RATIO $(\frac{P_{T2}}{P_A})$ = 1.00
4. GENERATOR ELECTRICAL LOAD = ZERO
5. PERCENT AIR BLEED $(\frac{W_{b1}}{W_A})$ = 0.5%
6. ROTOR SPEED = 324 RPM

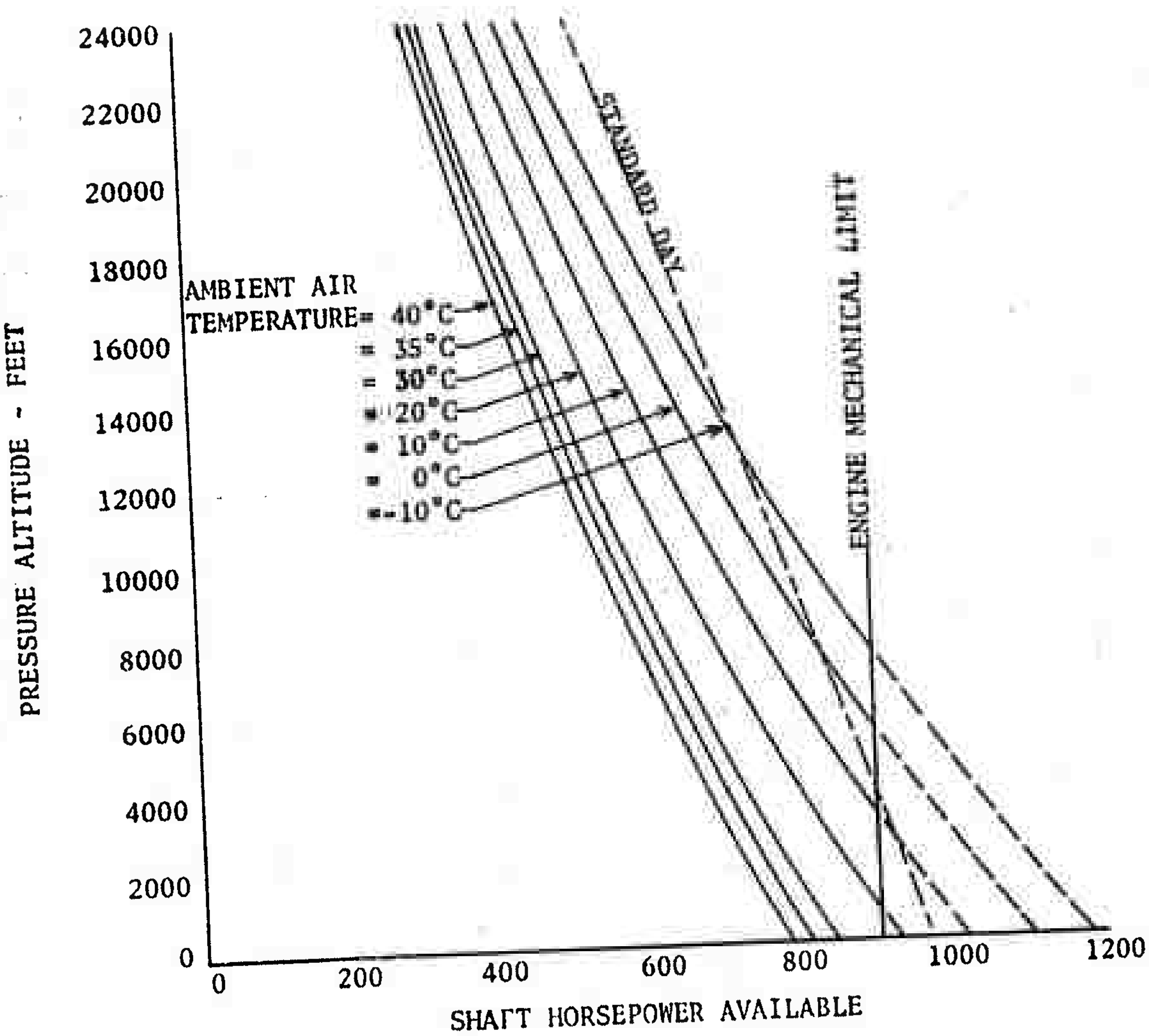
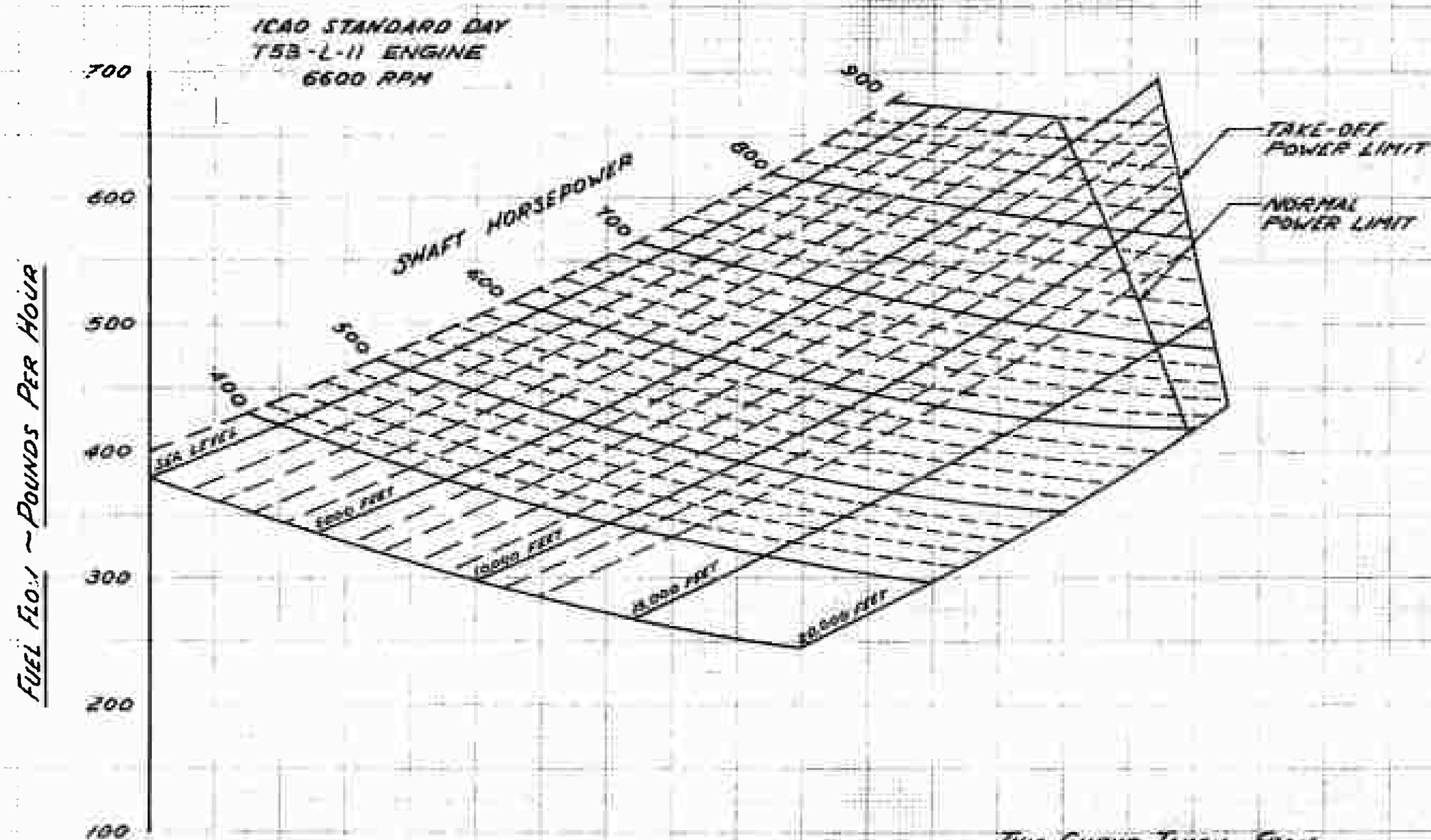


FIGURE NO. 76
SPECIFICATION FUEL FLOW



THIS CURVE TAKEN FROM
BELL REPORT 208-099-721

APPENDIX II. TEST METHODS AND DATA REDUCTION PROCEDURES

1.0 GENERAL

The equations and analysis method used to correct the performance of the helicopter to standard-day conditions are briefly described in this appendix.

The non-dimensional parameters used for data analysis are defined as follows:

$$C_P = \frac{550 \times SHP}{\rho A\ (\Omega R)^3} \text{ - Coefficient of Power}$$

$$C_T = \frac{W}{\rho A\ (\Omega R)^2} \text{ - Coefficient of Weight}$$

$$\mu = \frac{1.689\ V_T}{\Omega R} \text{ - Tip Speed Ratio}$$

where:

SHP = engine output shaft horsepower

ρ = air density, slugs/ft^3

A = total rotor disc area, ft^2

Ω = rotor angular velocity, radians/sec

R = rotor radius, ft

W = gross weight, lb

V_T = true velocity, kt

This non-dimensional method is useful only where blade stall or compressibility effects are not encountered.

1.1 POWER DETERMINATION

The T53-L-11 gas turbine engine incorporated a hydromechanical torquemeter as an integral part of the reduction gearing on the compressor end of the engine. This torquemeter was essentially a piston that supplied pressure, in proportion to the output torque, on the hydraulic oil contained in a cylinder. To obtain a more accurate indication of torque, the pressure of oil vapor behind this piston was also measured and the difference between this pressure and the hydraulic oil pressure was found. The conversion from torquemeter pressure to torque in inch-pound was obtained from the test cell run of engine S/N LE-10382.

The equation from which output shaft horsepower was determined from in-flight torquemeter and rotor rpm readings was derived as follows:

$$SHP = \frac{2\pi}{12 \times 33{,}000} \times N_E \times T$$

where:

SHP = output shaft horsepower

N_E = output shaft rotational speed, rpm

T = output shaft torque, in-lb

The torquemeter calibration as obtained from engine calibration data indicated that torque could be determined as the following function of torque pressure:

$$T = 219.0\ \Delta P$$

where:

ΔP = torque differential pressure, psi

Engine output shaft rotational speed was determined from rotor speed as follows:

$$N_E = N_R \times 20.37$$

where:

N_R = rotor rotational speed, rpm

Combining the above expressions resulted in the following expression for determining output shaft horsepower:

$$SHP = \frac{2\pi \times 219.0 \times 20.37 \times N_R \times \Delta P}{12 \times 33{,}000}$$

$$SHP = .07075 \times N_R \times \Delta P$$

Compressor inlet pressure or temperature instrumentation was not installed in the test aircraft. Since no pressure loss was noted in FTC-TDR-62-21 (reference z), the test ambient pressure was taken as the test compressor inlet pressure. Also, since the compressor inlet temperature noted in FTC-TDR-62-21 was consistently 2 degrees centigrade (C) above ambient, the test compressor inlet temperature was taken as 2 degrees C above the test ambient temperature.

1.2 LEVEL FLIGHT AND SPECIFIC RANGE

Level flight speed-power correction was derived from the C_P, C_T, and μ method. Each speed power was flown at a predetermined C_T holding rotor speed constant. To maintain W/ρ approximately constant, altitude was increased as fuel was consumed.

Test-day level flight power and airspeed data were corrected to standard-day conditions by the following method: The test-day speed-power point was defined by the dimensionless paramteres, C_{P_t}, C_{T_t} and μ_t. Correction of test-day power to standard-day conditions was made holding these coefficients constant on the standard day. It followed from this that the relationships below are true between test-day and standard-day conditions:

$$C_{P_t} = C_{P_s}, \quad C_{T_t} = C_{T_s}, \quad \mu_t = \mu_s$$

From these relationships and definitions of the particular terms the following relationships hold:

$$W_t/\rho_t = W_s/\rho_s, \quad \rho_s = \rho_{t_{avg}} \left[W_s/W_{t_{avg}}\right]$$

This last relationship permitted establishing the standard-day density, ρ_s, which was required for presenting the test-day data at a standard gross weight, W_s. W_s was the weight used in the computation of the C_T for each individual level flight test.

From the definition of the power coefficient, C_P, the following relationships could be derived:

$$SHP_t/\rho_t = SHP_s/\rho_s$$

$$SHP_s = SHP_t(\rho_s/\rho_t)$$

This relationships defined the standard-day power required for flying at the same thrust, power and speed coefficient as on the test day but under standard-day conditions. Each level-flight speed-power point was corrected in this fashion to standard-day conditions at the target gross weight.

Specific range calculations were performed using the level-flight performance curves presented in figures 1 through 6, appendix I, and the specification fuel-flow characteristics at 5-percent conservative presented in figure 76.

$$NAMPP = \frac{V_T}{W_f}$$

where:

$NAMPP$ = nautical air miles per pound of fuel

V_T = true airspeed in knots

N_f = fuel flow, pounds per hour

1.3 AUTOROTATIONAL DESCENTS

The observed rate of descent was corrected to a tapeline rate of descent by the following expression:

$$R/D = \frac{dhp}{dt} \left(\frac{T_t}{T_s}\right)$$

where:

$\frac{dhp}{dt}$ = slope of pressure altitude versus time curve at a given pressure altitude, fpm

T_t = test temperature at pressure altitude at which slope was taken - deg K

T_s = standard temperature for pressure altitude at which slope was taken - deg K

Data have been presented at test gross weight and test density altitude.

1.4 STABILITY AND CONTROL

The stability and control characteristics of the UH-1B/540 helicopter are discussed in terms of static stability, dynamic stability, and controllability. These terms are defined in the following paragraph.

1.4.1 Longitudinal Trim Curves

The longitudinal trim curves were determined from the position of the longitudinal cyclic control with respect to airspeed. The collective position was treated as an independent variable. For each test point, the collective-stick position was determined from the position normally used in flight. A longitudinal control position-airspeed gradient obtained from the trim curves determined apparent static stability. The stability is called apparent because it is an indication of the longitudinal static stability from the pilot's viewpoint but is not a direct measure of the speed stability or angle of attack stability of the aircraft. Longitudinal speed stability was obtained by locking the collective pitch at a trim point, then increasing or decreasing airspeed with the cyclic stick. Static lateral-directional stability was obtained by measuring control positions in steady-state sideslips. Control positions are reported in the following manner:

(a) Longitudinal and lateral cyclic displacement from full forward and full left. Full cyclic travel were 12.17 inches and 12.07 inches respectively.

(b) Pedal displacement in inches from a position with pedals aligned. Full travel was ± 3.5 inches.

(c) Collective pitch position in inches from full down. Full travel was 10.65 inches.

1.4.2 The dynamic stability of the helicopter was determined by recording aircraft behavior, displacement, rate and angular acceleration following an artificial disturbance. This artificial disturbance was the result of a pulse-type control input. The pulse input was made by rapidly displacing the control approximately 1 inch from trim position, holding for approximately 1 second, then rapidly returning to trim position and holding the control fixed. A mechanical fixture was used to guarantee precise input.

1.4.3 Controllability was treated in two parts: sensitivity and response. Sensitivity was defined as the maximum angular acceleration (degrees/second^2) of the aircraft per inch deflection of the cockpit control. Time to reach the maximum acceleration was included. Response was defined as the maximum angular velocity (degrees/second) of the aircraft per inch deflection of the cockpit control. Time to reach the maximum rate was included. The control deflections were stick-fixed, sudden, step-type inputs. The step input was made by rapidly displacing the control from trim and holding the control fixed until recovery was necessary. A mechanical fixture was used to insure precise inputs.

APPENDIX III. AIRCRAFT AND ARMAMENT DESCRIPTIONS

1.0 AIRCRAFT DIMENSIONS AND DESIGN DATA

a. Overall Dimensions

(1)	Aircraft length (nose to tail skid)	39.5 ft
(2)	Aircraft length (rotors turning)	52.9 ft
(3)	Width of skids	8.4 ft
(4)	Width (at horizontal stabilizer)	9.3 ft
(5)	Height (to top of turning tail rotor)	14.7 ft
(6)	Height (to top of rotor mast)	12.7 ft

b. Main Rotor

(1)	Number of blades	2
(2)	Rotor diameter	44 ft
(3)	Rotor solidity	0.0652
(4)	Disc area	1520 sq ft
(5)	Blade area (total)	99 sq ft
(6)	Blade chord (root to tip)	27 in
(7)	Blade airfoil (root to tip)	9-1/3 % Symmetrical Section Special
(8)	Blade twist	-10 deg
(9)	Flapping angle	$\pm$12 deg
(10)	Collective pitch angle limits (75% radius)	0 to 20 deg
(11)	Preconing angle	2-3/4 deg

c. Tail Rotor

(1)	Number of blades	2
(2)	Rotor diameter	8.5 ft
(3)	Rotor solidity	0.105
(4)	Disc area	56.7 sq ft
(5)	Blade area (total)	5.96 sq ft
(6)	Blade chord (root to tip)	8.41 in
(7)	Blade airfoil (root to tip)	NACA 0015
(8)	Blade twist	0 deg
(9)	Flapping angle	± 8 deg

d. Gear Ratios

(1)	Power turbine to engine output shaft	3.2057 to 1
(2)	Engine output shaft to main rotor	20.37 to 1
(3)	Engine output shaft to tail rotor	3.9902 to 1

e. Speeds

		Maximum	Minimum
(1)	Engine output shaft speeds	6600 rpm	6000 rpm
(2)	Main rotor speeds	323.8 rpm	294.5 rpm
(3)	Tail rotor speeds	1654.1 rpm	1503.7 rpm

2.0 POWER PLANT

The test aircraft was powered by a T53-L-11 gas turbine engine, S/N LE-10382. This engine is of the free-power turbine design, which consists of a two-stage planetary reduction gear section, five-stage axial and one-stage centrifugal compressor, diffuser, combustion chamber, first-stage turbine, second-stage turbine (free power), power shaft and an exhaust diffuser. The first-stage turbine drives the compressor and the second-stage turbine drives the power shaft. The power shaft extends coaxially through the compressor rotor and drives the reduction gearing at the forward end of the engine. Power for the main rotor is extracted through an internally-splined output gear shaft driven by the two-stage planetary reduction gearing. Power for the tail rotor is supplied from a takeoff on the lower end of the main rotor transmission. The engine has an output shaft operating range from 6000 rpm to 6600 rpm. The engine manufacturer's guaranteed power ratings are at 6610 rpm and standard-day sea-level conditions. The guaranteed ratings are 1100 shaft horsepower (shp) for takeoff power, 1000 shp for military power and 900 shp for normal rated power.

3.0 ROTOR AND CONTROL SYSTEMS

The 540 "door hinge" two-bladed, teetering, semi-rigid rotor system incorporates a flex beam hub by which the system attains a stiff chordwise or in-plane structure with a soft flapping or beam structure. A broad, flat steel plate replaces the standard UH-1B round hub spindle. This high in-plane stiffness permits the use of a large amount of tip weight without an increase in the chord oscillatory loads. The tip weight, in connection with the hub flexure, reduces the beam oscillatory load. This is intended to result in a dynamically balanced design which minimizes oscillatory stress levels and rotor-induced vibrations. The main rotor blade chord has been increased to 27 inches. The rotor remains at 44-feet diameter and features a 10-degree blade twist. The airfoil reaction is NACA 9-1/3 percent, which is thinner than the 12 percent used on all other UH-1 helicopters. Each blade has a 35-pound trim weight and a 20-pound weight installed in the leading edge "C" span section.

The rotating controls are similar to standard UH-1B/D controls except that they have been appropriately strengthened to resist the higher control loads encountered at the increased airspeeds and gross weight limits established for the UH-1B/540 aircraft. A change was made to the collective system which reverses the collective "A" frame pivot point and replaces the teflon bearings with needle bearings in certain main rotor rotating controls. The objective of this change was to eliminate the objectionable random 1-per-rev vibration that occurs after nominal service use.

4.0 TRANSMISSION ASSEMBLY

The transmission used in the UH-1B/540 helicopter is the same as the standard UH-1B transmission except for the quill assembly that drives the dual hydraulic system pumps

5.0 DUAL HYDRAULIC BOOST SYSTEM

The dual hydraulic boost system used in the UH-1B/540 aircraft is independent of the engine. Each system is completely independent of the other except that the hydraulic pumps are driven by the same transmission quill shaft. This system features independent dual reservoirs, pumps, tandem servo-actuators, filters, switches, valves, pressure indicators, and associated tubing and hydraulic lines. The pumps of both systems are powered by the main rotor. System No. 2 actuates the antitorque boost cylinder for tail rotor control. System No. 1 actuates any armament system requiring hydraulic power.

6.0 TAIL BOOM

The tail boom is the same as that of the standard UH-1B except for added camber on the trailing edge of the vertical fin and the incorporation of a UH-1D elevator with a protective shield on the leading edge.

7.0 TAIL ROTOR HUB AND BLADE ASSEMBLY

To withstand the higher tail rotor assembly loads normally encountered at higher airspeeds, a modified tail rotor hub assembly has been used for the UH-1B/540 aircraft. This hub is similar to the standard UH-1B/D aircraft hub except that the inboard bearing has been replaced by a thrust unit to reduce system chord loads.

8.0 AIRSPEED SYSTEM

The standard UH-1B airspeed system containing independent static and dynamic parts has been replaced by an integral static dynamic pitot tube, located on the cabin roof.

9.0 M-5 ARMAMENT SUBSYSTEM

The 40-millimeter (mm) M-5 grenade launcher helicopter armament subsystem as used on the UH-1B helicopters consists of two major components: the 40-mm M-75 grenade launcher and the grenade lanucher mount.

The turret assembly which is used to mount the grenade launcher is attached to three hard points outside the electronic equipment compartment and contains the components that mount, position and fire the M-75 grenade launcher.

The 40-mm M-75 grenade launcher is an air-cooled externally-powered, rapid-firing weapon capable of launching antipersonnel fragmentation-type projectiles. The M-75 is percussion fired and metallic-link-belt fed. It is mounted in the saddle assembly of the turret, which rotates on the horizontal axis to provide grenade-launcher elevation and depression. The saddle assembly in turn is mounted in the gimbal assembly, which rotates on the vertical axis to provide grenade-launcher left and right azimuth. Elevation and azimuth movements of the grenade launcher are made by the elevation and azimuth powered trunnion assemblies. Each powered trunnion assembly contains a direct-current drive motor which is powered by its respective servo-amplifier. The rotational travel of each powered trunnion assembly is limited by fixed mechanical stops and by adjustable limit-switch actuators. The grenade launcher as mounted on the UH-1B/540 helicopter has 60-degree left and right azimuth movement, 15-degree elevation and 35-degree depression.

An ejector chute assembly on the saddle assembly and an ejector chute on the gimbal assembly form a continuous chute for ejecting spent cartridge cases and misfired cartridges from the turret assembly.

The combination hand-control sight assembly (M-5/M-6) used with the XM-16 or XM-21 armament subsystem is also used for the M-5 armament subsystem. This sight provides the means for the aiming and firing of the M-75 grenade launcher by the copilot, who acts as the gunner. The sight assembly is so constructed and mounted that the relationship between the gunner's line of sight and the grenade launcher's line of fire is maintained throughout the field of fire.

10.0 XM-3 ARMAMENT SUBSYSTEM

The XM-3 is an armament subsystem designed to fire modified 2.75-inch limited-spin folding-fin aerial rockets (LSFFAR's) as an area weapon against personnel or soft targets. This system is capable of selective fire from the cabin, by either the pilot or copilot (gunner), in the following modes:

a. Pair, single rocket from each launcher.

b. Ripples of 1-2-3-4-6 or 24 pairs at 6 pairs per second.

The 2.75-inch rocket fire control system provides a means of firing rockets from launchers on both sides of the aircraft.

The launchers are of the open-breech tube type. Each launcher consists of 4 modules containing 6 tubes each. Except for the trunnion shaft, bearings and fittings, the launchers are constructed of aluminum and their support structures attach to hard points on the underside of the aircraft. The launchers have manual mechanical adjustments from +6 degrees to -6 degrees relative to the waterline of the helicopter. They can be jettisoned by means of explosive bolts. Two of these explosive bolts, one near each end of the launcher crank, are used per launcher to attach the launcher's adapter frame to the crank. The explosive bolts are the double-shear type, consisting of a shear bolt, bushing, piston, shear pin, sealing ring, base, power cartridge and collar. A switch on the rocket armament panel, accessible to the pilot and copilot, can detonate the bolts and jettison the launchers.

The junction box is a flat rectangular-shaped aluminum chassis attached to brackets, located in the baggage compartment on the right side of the aircraft. The junction box contains the circuitry necessary for firing the weapons and for jettisoning the two pod assemblies. The junction box and rocket armament panel are electrically connected by connectors located on the rear of the panel assembly. The controls necessary for jettisoning the two pod assemblies, for turning on system power, and for selecting the number of pairs of rounds in a ripple are located on the front of the rocket armament panel. Also located on the front of this panel are a counter to record the number of pairs of rounds fired during the mission, a zero indicator light, an "armed" indicator light, a system "power-on" indicator light, a "safe" indicator light, a jettison "power-on" indicator light, and a jettison "complete" indicator light.

The Mark 8 sight assembly consists of a Mark 8 collimating-reflector-type sight which contains an illuminated adjustable reticle pattern. The sight is mounted from the instrument panel by means of a bracket mounted directly over the helicopter altimeter. The sight adjustable mechanism allows the line of sight to be elevated or depressed +15 or -15 degrees from zero elevation. The sight is illuminated by a reticle control located in the instrument panel pedestal.

11.0 XM-16 ARMAMENT SUBSYSTEM

The XM-16 armament subsystem is composed of a combination of the M-6 subsystem and the 7-round, 2.75-inch LSFFAR XM-157 rocket pod. The M-6 subsystem consists of four 7.62-mm M-60C machine guns, two machine-gun mount assemblies, and the necessary controls and hardware. A gun mount is attached to the rack assembly of the external stores support assembly on each side of the

UH-1B helicopter, and the guns are aimed by means of a sighting station at the copilot's position. The four machine guns have a total weight of 796 pounds and a maximum capacity of 6600 rounds. Total lateral deflection is 12 degrees inboard to 70 degrees outboard and the vertical deflection is 9 degrees upward to 66 degrees downward. When either set of guns is traversed to its inboard limit stop, the guns cease firing.

The control panel consists of an OFF-SAFE-ARMED switch and a gun selector switch. Both switches are three position toggle type and must be pulled upward to be operated. The guns fire only when the OFF-SAFE-ARMED switch is in the ARMED position. The gun selector switch enables the operator to select his fire power: with the switch in the LOWER position, only the lower guns operate; in the ALL position, both upper and lower guns operate; and in the UPPER position, only the upper guns operate.

The sighting station is located at the copilot's position and provides the means for remotely aiming the guns. When the "dead-man"switch is depressed, control of the guns is transferred from the cyclic control stick firing switches to the controller trigger. In this condition the guns cannot be fired from the cyclic control stick. The movement of the controller in elevation and deflection causes the guns to follow the controller. When the "dead-man" switch is released, the guns are returned to the "stow" position and can then be fired by depressing the fire button on the pilot's cyclic control stick.

Two 7-round, 2.75-inch LSFFAR XM-157 rocket launcher pods, one for each side of the helicopter, are suspended from the MA-4A bomb racks and are expendable. The launcher attitude is of fixed variable design and can be changed in elevation only from the ground. The 2.75-inch LSFFAR's can be fired in ripples only and are ignited in pairs, one rocket simultaneously from each launcher. Up to 7 pairs of rockets may be selected. The number of pairs of rockets may be preset before firing and the subsystem is capable of firing 7 pairs of rockets forward from a fixed position of the launcher in 1.167 seconds. The rockets are fired from the same cyclic control stick firing switches that are used by the pilot or copilot to fire the M-6 machine guns. The trigger switches are located on each control stick. The rockets are aimed by pointing the aircraft and using either the M-6 machine guns as spotter rounds or the Mark 8 sight mounted in the pilot's position. The two rocket pods can be jettisoned simultaneously by either electrical or mechanical means.

The machine guns and rockets cannot be fired simultaneously. When the ROCKET-GUN switch is placed in ROCKET, the copilot may use the controller on the sighting station to aim and fire the guns. The instant the pilot depresses the trigger on the cyclic control stick the guns automatically stop firing and the rockets are ignited.

12.0 XM-21 ARMAMENT SUBSYSTEM

The XM-21 armament subsystem consists of a combination of 7.62-mm twin, high-rate-of-fire XM-20 machine guns and twin 2.75-inch XM-157 rocket launchers. The gun mount assemblies, which are installed one on each side of the helicopter, were originally designed to support two M-60C machine guns. Each mount of the test aircraft was modified to install a single, recoil-mounted. automatic machine gun.

The XM-20 is an electrically-driven, 6-barrel, Gatling-type, high-rate-of-fire machine gun. The two guns weigh 100 pounds. The weapon is capable of providing fire coverage up to 10 degrees in elevation, 85 degrees in depression, 12 degrees inboard, and 70 degrees outboard at rates of 2000 to 4000 rounds/minute. As with the M-6 subsystem the guns cease firing when either weapon traverses to its inboard limit. The slew rates are 40 degrees/second in elevation and depression and 75 degrees/second in deflection.

The sighting station, which is located at the copilot's position, is identical to the M-6 subsystem sighting station; and the operational functions are the same. When the "dead-man" switch is depressed, the gun turrets follow the action of the controller on the sighting station and the guns can be fired only by the copilot. With the release of the "dead-man" switch, the guns return to the "stow" position, and both the pilot and copilot can fire the guns from the cyclic control sticks. The pilot is capable of firing the guns only in the "stow" position and directs the fire by aiming the helicopter.

Each of the XM-20 weapons is fed through a flexible ammunition chute supported at the forward side of the pylon. The rounds are fed from the right side, and the spent cases are ejected rearward and to the left. The links are ejected rearward and to the right by means of a rotary-type delinking feeder. The ammunition storage box configuration in the aircraft remains the same as that of the M-6 subsystem. Two forward rows of boxes supply the left-hand gun and two aft rows supply the right-hand gun. There is a total of 3000 rounds for each gun which are linked together to produce a single continuous belt through the cartridge drive crossover. The cartridge drive crossover enables each gun to be fed from the two rows of storage boxes at the dual rate of 2000 and 4000 rounds/minute. There is a burst limit time delay of approximately 3 seconds in the firing system.

The control panel is very similar to that of the M-6 subsystem except for the gun selector switch. The operator has the choice of firing either the left-hand gun only, the right-hand gun only, or both guns simultaneously.

The rocket launcher is the same 7-round, 2.75-inch XM-157 LSFFAR rocket pod as that of the XM-16 armament subsystem. The capabilities and firing sequences are also identical. Rocket firing is primary with the ROCKET-GUN switch in the ROCKET position. The number of rocket pairs to be fired per burst is selected on the ROCKET PAIR SELECTOR switch. Depressing a cyclic control stick trigger causes the preselected number of rocket pairs to fire. The circuitry is reset to the original condition whenever the trigger is released during a rocket burst so that the full selected number of rocket pairs will be fired at the next burst. Should the copilot be firing machine guns with the sighting station, depressing the pilot's cyclic control stick trigger stops the machine-gun fire and causes the rocket pairs to be fired. The pilot directs the rocket fire by maneuvering the aircraft and acquires the target through his reflex sight.

13.0 WEIGHT AND BALANCE

The weight and balance evaluation of the test aircraft was conducted in a closed hangar by USAAVNTA personnel. The weights of the aircraft with the various armament subsystems installed are categorized below:

Item	XM-16/M-5 Armament Subsystems	Weight lb
1	Basic weight	4950
2	Crew of 2 @ 200 lb each	400
3	242 gallons of fuel @ 6.5 lb/gal	1573
4	Armored seats (2)	300
5	M-5 armament subsystem with associated external and internal components and 150 rounds of 44-mm ammunition	340
6.	XM-16 armament subsystem with associated external and internal components and 6000 rounds of 7.62-mm ammunition and 14 2.75-inch rockets	1101
	Engine Start Gross Weight	8664

XM-21/M-5 Armament Subsystems	Weight lb
Items 1 through 5 of the XM-16/M-5 armament subsystem are identical	7563
Item 6 is replaced by the XM-21 armament subsystem	1108
Engine Start Gross Weight	8671

XM-3/M-5 Armament Subsystems	Weight lb
Items 1 through 5 of the XM-16/M-5 armament subsystem are identical	7563
Item 6 is replaced by the XM-3 armament subsystem with 48.2.75-inch rockets	1338
Engine Start Gross Weight	8901

APPENDIX IV. INSTRUMENTATION

1.0 Calibrated instruments were installed and maintained by USAAVNTA. The following parameters were recorded:

a. PILOT'S PANEL

(1) Sensitive Rotor Speed

(2) Boom System Airspeed

(3) Boom System Altitude

(4) Angle of Sideslip

(5) Angle of Attack

(6) Longitudinal Cyclic Stick Position

(7) Lateral Cyclic Stick Position

(8) Collective Stick Position

(9) Pedal Position

b. ENGINEER'S PANEL

(1) Standard System Airspeed

(2) Standard System Altitude

(3) Torque (High and Low)

(4) Free Air Temperature

(5) Fuel Flow (Stepper Motor System)

(6) Fuel Total

(7) "G" Forces

(8) Oscillograph Count

c. OSCILLOGRAPH

(1) Pilot's Event

(2) Engineer's Event

(3) Longitudinal Cyclic Stick Position

(4) Lateral Cyclic Stick Position

(5) Pedal Position

(6) Collective Stick Position

(7) Rotor Blip

(8) C.G. Normal

(9) Angle of Attack

(10) Angle of Sideslip

(11) Pitch Angle

(12) Pitch Rate

(13) Pitch Acceleration

(14) Yaw Angle

(15) Yaw Rate

(16) Yaw Acceleration

(17) Roll Angle

(18) Roll Rate

(19) Roll Acceleration

(20) Pilot's Vertical Vibration

(21) Pilot's Lateral Vibration

(22) Aft Bulkhead Vertical Vibration

(23) Aft Bulkhead Lateral Vibration

(24) Voltage

(25) Linear Rotor RPM

2.0 In addition to the parameters listed, pickups that correlated the firing time of the explosive bolts during jettison tests were installed. During the rates-of-fire tests of the various armament subsystems the oscillograph was used to record the rounds being fired.

APPENDIX V. SYMBOLS AND ABBREVIATIONS

Symbols and Abbreviations	Definition	Units
A	Rotor Disc Area	ft^2
C.G.	Center of Gravity	in
C_P	Power Coefficient	Non-dimensional
C_T	Thrust Coefficient	Non-dimensional
$\frac{dh_p}{dt}$	Slope of Pressure Altitude versus Time Plot	--
freq	Frequency	cycles/sec
IGE	In Ground Effect	--
KCAS	Knots Calibrated Airspeed	kt
KIAS	Knots Indicated Airspeed	kt
KTAS	Knots True Airspeed	kt
NAMPP	Nautical Air Miles per Pound of Fuel	--
N_R	Rotor Rotational Speed	rpm
N_E	Engine Output Shaft Rotational Speed	rpm
R	Rotor Radius	ft
R/D	Rate of Descent	fpm
rd/min	Rounds per Minute	--
RPM/rpm	Revolution per Minute	rpm
S.A.	Single Amplitude	g
SHP/shp	Shaft Horsepower	ft-lb/min

Symbols and Abbreviations	Definition	Units
T	Output Shaft Torque	in-lb
ΔP	Torque Differential Pressure	psi
V_{NE}	Airspeed Not to Exceed	kt
V_T	True Airspeed	kt
W	Gross Weight	lb
W_f	Fuel Flow	lb/hr
1/rev	Cycles per Rotor Revolution (vibrations)	cycles/rev
δ	Pressure Ratio	--
μ	Rotor Tip Speed	Non-dimensional
Ω	Angular Velocity	radians/sec
ρ	Air Density	slugs/ft^3
θ	Temperature Ratio	--
Subscript		
s	Standard-Day Conditions	
t	Test Conditions	

APPENDIX VI. REFERENCES

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11 SUPPLEMENTARY NOTES	12 SPONSORING MILITARY ACTIVITY
	Iroquois Project Manager U.S. Army Materiel Command

13 ABSTRACT This report presents the results of an engineering flight test of the UH-1B/540 rotor helicopter equipped with the XM-16, XM-21, or XM-3 armament subsystem in conjunction with the M-5 armament subsystem. The test was conducted by the U.S. Army Aviation Test Activity (USAAVNTA). Overall test objectives were to verify safety of flight, develop data for the operator's manual, and assure that aircraft modifications did not degrade the handling qualities or limit the operational characteristics of the subsystems. Specific objectives were to determine quantitatively the effect of the armament subsystems on stability, control and performance of the aircraft, to determine the rocket launcher jettison characteristics and to define the usable limits of the flight envelope for safe jettison of the launchers. Testing was conducted at Edwards Air Force Base, California, and at sites in Fort Irwin and Bakersfield, California. A total of 152 flights for a productive flight time of 116.4 hours was flown on aircraft S/N 64-14105 between 13 November 1965 and 5 May 1966. This included 30 jettison flights and 35 firing flights. There were no significant adverse changes in the stability and control characteristics of the UH-1B/540 helicopter due to the installation of the various armament subsystems. A drag penalty imposed by the installation of the XM-3/M-5 or XM-21/M-5 caused a 13-percent and 10-percent decrease in specific range respectively with a corresponding 20-percent and 11-percent decrease in airspeed. The vibration level of the aircraft was generally satisfactory. A self-excited undamped lateral 2/3 per-rev vibration grounded the aircraft and terminated testing. This characteristic could have safety-of-flight implications and should be corrected. Firing the various armament subsystems could be conducted at all airspeeds within the flight envelope with no major stability and control problem encountered. Firing rockets in a hover, with the launcher at negative deflection, should be avoided. Rocket launcher jettison can be satisfactorily accomplished under all level flight conditions tested. Jettison should be avoided during autorotations and in close proximity to the ground.

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